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CONSTRUCTION AND DESIGN OF ROCKET ENGINES.(U)

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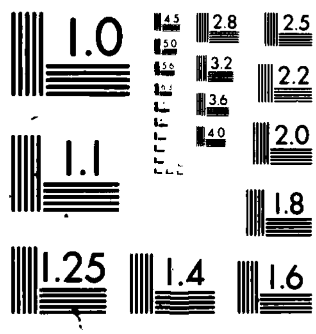

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FOREIGN TECHNOLOGY DIVISION

AIR FORCE SYSTEMS COMMAND

CONSTRUCTION AND DESIGN OF ROCKET ENGINES

by

V. A. Volodin

FOREIGN TECHNOLOGY DIVISION

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TABLE OF CONTENTS

U. S. BOARD ON GEOGRAPHIC NAMES TRANSLITERATION SYSTEM

Block	Italic	Transliteration	Block	Italic	Transliteration
А а	*А а*	A, a	Р р	*Р р*	R, r
Б б	*Б б*	B, b	С с	*С с*	S, s
В в	*В в*	V, v	Т т	*Т т*	T, t
Г г	*Г г*	G, g	У у	*У у*	U, u
Д д	*Д д*	D, d	Ф ф	*Ф ф*	F, f
Е е	*Е е*	Ye, ye; E, e*	Х х	*Х х*	Kh, kh
Ж ж	*Ж ж*	Zh, zh	Ц ц	*Ц ц*	Ts, ts
З з	*З з*	Z, z	Ч ч	*Ч ч*	Ch, ch
И и	*И и*	I, i	Ш ш	*Ш ш*	Sh, sh
Й й	*Й й*	Y, y	Щ щ	*Щ щ*	Shch, shch
К к	*К к*	K, k	Ъ ъ	*Ъ ъ*	"
Л л	*Л л*	L, l	Ы ы	*Ы ы*	Y, y
М м	*М м*	M, m	Ь ь	*Ь ь*	'
Н н	*Н н*	N, n	Э э	*Э э*	E, e
О о	*О о*	O, o	Ю ю	*Ю ю*	Yu, yu
П п	*П п*	P, p	Я я	*Я я*	Ya, ya

*<u>ye</u> initially, after vowels, and after ъ, ь; <u>e</u> elsewhere.
When written as ё in Russian, transliterate as yё or ё.

RUSSIAN AND ENGLISH TRIGONOMETRIC FUNCTIONS

Russian	English	Russian	English	Russian	English
sin	sin	sh	sinh	arc sh	$\sinh^{-1}$
cos	cos	ch	cosh	arc ch	$\cosh^{-1}$
tg	tan	th	tanh	arc th	$\tanh^{-1}$
ctg	cot	cth	coth	arc cth	$\coth^{-1}$
sec	sec	sch	sech	arc sch	sech^{-1}
cosec	csc	csch	csch	arc csch	csch^{-1}

Russian	English
rot	curl
lg	log

CONSTRUCTION AND DESIGN OF ROCKET ENGINES.

V. A. Volodin.

Page 1.

It is approved by the Ministry of the Higher and secondary special education of the USSR as the textbook for the technical schools.

Page 2.

In the textbook are given common survey/coverage, the classification and short rocket-motor characteristics and their working medium/propellants. Is briefly presented the history of the development of rocket engines. Is examined the theory of thermal rocket engines and are presented the bases of construction and design of the rocket engines, which work on the liquid and solid chemical propellant. Is given some information about the nuclear and electrical rocket engines.

Textbook is intended for the studying machine-building technical schools. It can be useful to the technical-engineering workers of rocket engine construction.

Page 3.

Preface.

In last 10-15 years is vigorously developed rocket and space technology. Are created different types of the highly efficient rocket engines, which work on the liquid and solid chemical propellant, are developed/processed nuclear and electric motors. In a number of cases in one rocket apparatus simultaneously are used the engines of different types.

At present there is a sufficiently numerous literature according to the theory, by the construction/design and according to the design of RD. However, until recently there is no textbook on the rocket engines for the technical schools. Target of this book - to complete this gap/spacing.

Textbook consists of four parts.

In the first part are given the general information about the

rocket engines and the rocket apparatuses (principle of their effect/action, classification, the parameters, the field of application, etc.). Furthermore, are given the classification of working medium/propellants of RD and the requirements, presented to them.

In the second part from the position of the generalized concept "thermal rocket engine" is examined the theory chemical and nonchemical RD.

The third part of the textbook is indicated to construction/design and design of the chemical rocket engines (first of all of liquid ones), which at present underwent great development.

In one fourth is given some information about nonchemical RD.

All parameters in the textbook are given in the international system of units (to system of SI). The figures, given in the book, for the educational goals are carried out simplified.

The author expresses deep appreciation to doctor of technical sciences, Prof. G. B. Sinyarev to doctor of technical sciences M. R. Gnesin for the considerable attention to the publication of present textbook, and also for the valuable observations whose account made

it possible to considerably improve his quality.

Page 4.

During the review of the manuscript of the book G. B. Sinyarev communicated to the author the number of original materials according to the classification, the methods of calculation and comparison of the engines of different types. These observations and materials were used for the work above the book.

During the writing of some chapters of textbook were useful the observations of eng. N. V. Ivanov and Cand. of tech. sciences Ye. A. Ivan'kova. Great assistance the author they showed/rendered with accomplishing of calculations of eng. D. T. Volodin, and in the formulation of the figures - eng. T. I. Ivanov. By all to them the author expresses his gratitude.

The author with the appreciation will take critical observations and wishes about an improvement in the books which should be sent to an address: Moscow, B-66, 1st Basmannyy st., 3, publishing house "Mashinostroyeniye".

Page 5.

Part I.

ROCKET ENGINES AND ROCKET APPARATUSES.

Chapter 1.

GENERAL INFORMATION ABOUT THE ROCKET ENGINES.

§ 1.1. Reaction forces. General concepts about jet engines.

Jet engine is called the engine which creates force for displacing the apparatus in the path space the energy conversion of its own or external source into the kinetic energy of the stream of substance. For the work of jet engine can be used both the substance, placed on board the apparatus, and environment, i.e., the medium in which works the engine. The stream of the substance, which escapes from the jet engine, is called exhaust jet, and the force which appears as a result of its outflow - reaction force.

The parameters and the state of aggregation of substance before the utilization in the engine (i.e., parent substance) and in exhaust

jet usually substantially are distinguished.

Parent substance of exhaust jet can be found in the gaseous, liquid or solid state and have a temperature, equal to ambient temperature. Exhaust jet is most frequently the high-temperature mixture of gases. The type of the utilized substance in many respects is determined by the type of jet engine.

One of the basic components of any jet engine is chamber/camera; in its initial part the substance is shifted into the state which it must have into exhaust jet. For example, in the series/number of jet engines liquid chemical propellant is supplied into the chamber/camera, in initial part of which (in the combustion chamber) it burns, isolating heat and forming gaseous products. The final part of the chamber/camera, called nozzle, serves for accelerating the combustion products.

Page 6.

The jet engines, utilized at present, are the broad class of the engines of the most varied designation/purpose. The region of their use/application constantly is expanded.

For the creation of reaction force they are necessary:

a) the substance which in the form of exhaust jet is thrown out from the engine; the substance indicated we will subsequently call working medium/propellant;

b) the source of primary energy, which is converted into the kinetic jet energy;

c) engine, i.e., the device, which ensures the conversion indicated.

By working medium/propellant or its composite/compound component part they can be:

a) the gaseous or liquid environment, for example the atmosphere of the Earth and other planets or water;

b) the substance, placed in the special capacities (tanks) of apparatus or is direct in the engine chamber;

c) the mixture of the environment (for example, air) and the substance (for example, kerosene), supplied to the chamber/camera from the tank of the apparatus (this substance can be also placed

directly in the chamber/camera).

Primary energy is stocked on board the apparatus in any source or is accepted from the external source (for example, from the sun).

To the smallest degree depend on the environment the jet engines, in which is working the body and the source of primary energy they are placed on the apparatus itself. They are used extensively on the flight vehicles, called rockets; therefore such jet engines isolate into the separate class of rocket engines (RD). Rocket engine is the only type of the engine which can work in any gaseous and liquid medium, and also in the vacuum (vacuum).

As a result of the outflow or exhaust jet from the rocket engine the mass of apparatus in the majority of the cases rapidly is changed. Therefore rocket-propelled device is the body of variable mass.

Rocket engines differ significantly from the systems which consist of engine and motor, which develops reaction force. Example is the system "aviation gas-turbine engine+propeller", in which the motor (propeller), given by engine, accelerates the incident by it airflow, as a result of which is created the reaction force. The processes, which occur in the engine and the motor, and also the

working medium/propellants, which take place through them, to a considerable degree differ from each other. Therefore this system is called the engine of indirect reaction.

Rocket engines are united into the unit strictly engine (for example, combustion chamber) and motor (for example, nozzle).

Page 7.

Reaction force is created as a result of increasing motion of working medium/propellant, moreover is working the body, which takes place through the engine and the motor, one and the same. Therefore rocket engines are called also the engines of forward reaction.

§ 1.2. Basic types of rocket engines.

Before examining the classification and the basic parameters of rocket engines, let us dismantle/select the simplest diagrams and the operating principle of most characteristic ones of them. Let us introduce the concept about the engine installation (DU), which encompasses the source of primary energy, tank with the working medium/propellant and engine.

DU with the liquid propellant rocket engine (ZhRD). As the

working medium/propellant in ZhRD most frequently are used two liquid specially selected substances: oxidizer and combustible. Oxidizer and fuel, interacting with each other, call chemical fuel/propellant, or it is simple by fuel/propellant. Oxidizer and combustible are propellant components.

With the course of the reaction of burning in the combustion chamber occurs conversion of primary (chemical) energy of fuel/propellant into the heat, as a result of which are formed the combustion gases, which have high temperature. The acceleration of combustion products in the nozzle of chamber/camera as a result of the conversion of their heat into the kinetic energy leads to the creation of reaction force.

Fig. 1.1 depicts the simplest diagram of DU with ZhRD. Installation consists of chamber/camera 1, fuel tank 3, of oxidizer tank 6, of tank/balloon with compressed gas 4 and valves 2, 5 and 7. Compressed gas during valve opening 5 enters from the tank/balloon into the tanks, as a result of which the pressure in them increases/grows. During valve opening 2 and 7 fuel and oxidizer begin to enter chamber/camera and in it begins the process of burning.

Solid-propellant rocket engine (RDTT). The rocket engine, which works on the solid fuel, also relates to the chemical engines. Solid

chemical fuel/propellant is the finished mixture of oxidizer and fuel which in the form of charge is placed directly in the combustion chamber. If the process of burning in RDTT has some differences from the process of burning in ZhRD, then the processes of expanding the combustion products in the nozzles of RDTT and ZhRD in many respects are analogous.

Fig. 1.2 shows the simplest diagram of RDTT. It consists of housing 2, nozzle 4, solid-propellant grain 3 and igniter 1. During the supplying of command/crew to the igniter occurs the ignition of the solid-propellant grain which burns from the surface and are formed the products of combustion, which escape behind the nozzle.

DU with the Nuclear rocket engine (YaRD). The energy source in the nuclear rocket engines are the reactions, which take place with a change in the nuclear structure, including of fission reaction of the nuclei of substance with the large atomic mass (for example, the isotopes of uranium); in the process of these reactions is isolated a large quantity of heat.

Page 8.

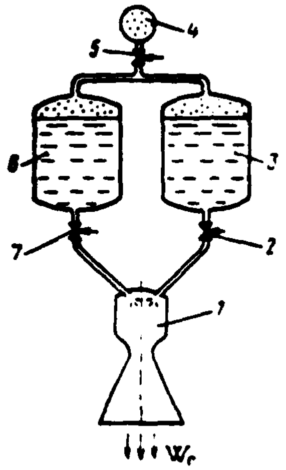
Fig. 1.3 depicts the simplest diagram of DU with YaRD. Installation consists of nuclear reactor 1, tank 3 with the working

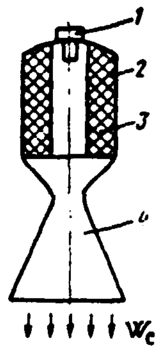
medium/propellant, pump 5, valve 4 and turbine 2. Working body from the tank is pumped into nuclear reactor in which is placed the fissionable material. Flowing/occurring/lasting through the reactor, working body vaporizes and is heated to the high temperature by the heat, isolated during the nuclear fission of fissionable material. The gaseous products of the vaporization of working medium/propellant flow out behind the nozzle, creating thrust.

The power, necessary for the work of pump, is transmitted to it through the shaft from the turbine. In turn, for the work of turbine to it is supplied certain quantity of gaseous products of the vaporization of working medium/propellant, selected/taken behind the nozzle.

Engines examined above have the general/common/total special feature/peculiarity, consisting in the fact that in them primary (chemical or nuclear) energy is converted first into the heat, and then into the kinetic jet energy. Therefore ZhRD, RDTT and YaRD relate to the class of thermal rocket engines.

DU with the electrical rocket engine (ERD).





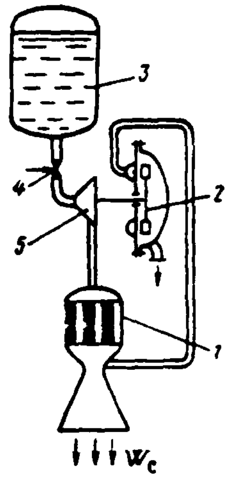


Fig. 1.1. Fig. 1.2. Fig. 1.3.

Fig. 1.1. Simplest diagram of DU with ZhRD: 1 - chamber/camera, 2, 5, 7 - valves; 3 - fuel tank; 4 - tank/balloon with compressed gas; 6 - oxidizer tank.

Fig. 1.2. Simplest diagram of RDTT: 1 - igniter; 2 - housing; 3 - solid-propellant grain; 4 - nozzle.

Fig. 1.3. Diagram of DU with YaRD: 1 - nuclear reactor; 2 - turbine; 3 - tank with working medium/propellant; 4 - valve; 5 - pump.

Page 9.

Electrical rocket engines differ significantly from the thermal by the fact that for dispersing/accelerating the working medium/propellant in the motor electrical RD is used electrostatic or electromagnetic field; with its help the electrical energy is converted into the kinetic jet energy. Therefore working body electrical RD must possess in exhaust jet the specific electrical properties.

For producing the electrical energy, utilized for accelerating the working medium/propellant, necessary is the source of primary energy.

Basic units of DU with ERD and nuclear source of primary energy (Fig. 1.4) are nuclear power plant 7, tank 3 with the working medium/propellant, device 5 for the creation of electrically charged particles and device 6 for their dispersal/acceleration with the aid of electrostatic or electromagnetic field (motor). The discharge velocity of working medium/propellant behind the nozzle electrical RD is considerably (by an order) more than in thermal ones.

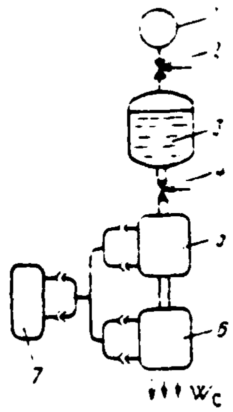


Fig. 1.4. The simplest diagram of DU with ERD: 1 - tank/balloon with the compressed gas; 2, 4 - valves; 3 - tank with the working medium/propellant; 5 - device for the creation of electrically charged particles; 6 - accelerating device of electrically charged particles (motor); 7 - nuclear power plant.

§ 1.3. Rocket engine thrust.

Thrust of RD - this is the force with which the engine acts on rocket apparatus, causing its displacement/movement in the space, or to the stand on which is established/installed the engine.

Rocket engine thrust in the vacuum. Let us first examine the case of the work of engine in the vacuum (out of the limits of the atmosphere of the Earth and other planets). Such conditions are characteristic for RD of space vehicles (LA) and many rockets.

Concept "vacuum" can be connected with upper border of atmosphere, namely, it is possible to consider that with an increase in altitude of flight of LA the pressure of the atmosphere becomes equal to zero when aerodynamic resisting forces are decreased to the negligible values.

Let us examine the derivation of the formula of thrust in the vacuum based on the example of rocket apparatus with ZhRD (Fig. 1.5). Thrust in the vacuum let us designate $P_п$. In order to simplify the derivation of the formula of thrust RD, let us accept the following assumptions.

1. Flow of gas during motion along chamber/camera unidimensional, i.e., gas moves in parallel to axis/axle of chamber/camera, moreover in each cross section of chamber/camera gas velocity with respect to entire cross section is identical. Actually gas velocity over the cross section is not strictly identical and, furthermore, gas in the nozzle moves not only in the axial, but partly, due to the complex nozzle configuration, and in the radial direction (Fig. 1.6).

Page 10.

2. Flow of gas in chamber/camera being steady (stationary), i.e., parameters of gas in each of its cross sections do not change in the course of time.

3. Negligible by speed of motion of liquid propellant in tank W along its axis/axle (speed indicated is low).

Let us accept the direction, opposite to the direction of the motion of gas, for the positive. Let us examine the forces, which effect on the part of the gas flow, which is located within the chamber/camera. Such forces two (Fig. 1.7):

a) the force of solid chamber walls. This force is equal in magnitude to the interesting us reaction force of P and is opposite to it in the direction, i.e., it is equal to -P;

b) the force of gas flow, which is located beyond nozzle exit section (this cross section is called also nozzle edge). The parameters, which relate to nozzle exit section, we will designate by index "c":

p_c - gas pressure in nozzle exit section;

f_c - nozzle exit area;

W_c - gas velocity in the cross section indicated, etc.

The force in question is equal to product $f_c p_c$; it is directed to the side, opposite to the direction of the motion of gas.

During the design of the forces examined to the longitudinal axis of chamber/camera we obtain composite force P_Σ, effecting on the gas flow, flowing on the chamber/camera,

$$P_\Sigma = -P + f_c p_c. \qquad (1.1)$$

Let us introduce the following designations:

1) $m_{нач}$ - initial mass of flight vehicle (to the start of engine and of the start of apparatus);

2) $m_{кон}$ - the finite mass of apparatus (after engine cutoff).

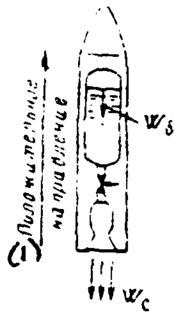


Fig. 1.5.

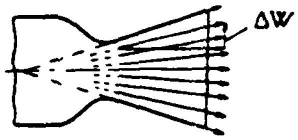


Fig. 1.6.

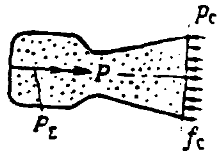


Fig. 1.7.

Fig. 1.5. To conclusion/derivation of equation of rocket engine thrust (based on example of rocket apparatus with ZhRD).

Key: (1). Positive direction.

Fig. 1.6. Arrangement of gas jets, which escape/ensue from nozzle: ΔW - speed loss to nonparallelism of gas jet of axis/axle of nozzle.

Fig. 1.7. Forces, which effect on flow of gas, which is located within chamber/camera.

Page 11.

Value $\Delta m = m_{нач} - m_{кон}$ is equal to the mass of fuel/propellant or reaction products, spent by engine for the time of its work.

A change in the momentum of mass Δm equally (let us consider the sense of the vector of speed $W_{\text{б}}$ and W_c)

$$\Delta m[(-W_c)-(-W_{\text{б}})]$$

or $\Delta m(W_c-W_{\text{б}})$.

According to theorem of momentum the power impulse, which arose for the time $\Delta\tau$ as a result of the acceleration of the rejected mass, is equal to a change in the quantity of its motion.

Consequently,

$$P_z\Delta\tau=-\Delta m(W_c-W_{\text{б}}).$$

Taking into account assumption $W_{\text{б}}=0$, we obtain

$$P_z=-\frac{\Delta m}{\Delta\tau}W_c$$

or taking into account equation (1.1)

$$-P+f_c p_c=-\frac{\Delta m}{\Delta\tau}W_c.$$

Hence $P=\frac{\Delta m}{\Delta\tau}W_c+f_c p_c$.

For the case of work of RD in vacuum $P=P_{\text{п}}$. Therefore $P_{\text{п}}=\frac{\Delta m}{\Delta\tau}W_c+f_c p_c$.

With the work of engine on invariable mode

$$\frac{\Delta m}{\Delta\tau}=\dot{m}=\text{const},$$

where m - mass [1] flow rate for the time unit (1 s).

FOOTNOTE [1]. In the textbooks and other technical books, created without taking into account the system of SI, frequently is used the weight flow rate per second, moreover it they designate by letter G. In the overwhelming majority of the cases it is more convenient instead of the weight ones to use the mass parameters as a result of simplification in equations and independence of mass from the forces of gravitational fields. ENDFOOTNOTE.

Consequently, rocket engine thrust in the vacuum

$$P_{п}=\dot{m}W_{c}+f_{c}p_{c}. \quad (1.2)$$

Page 12.

For the series/number of calculations it is convenient to express thrust in the vacuum more simply:

$$P_{п}=\dot{m}W'_{э.п}, \quad (1.3)$$

where $W_{э.п}$ - effective escape velocity. Value $W_{э.п}$ can be determined, after substituting expression (1.3) into equality (1.2)

$$\dot{m}W'_{c}+f_{c}p_{c}=\dot{m}W_{э.п}.$$

Hence

$$W'_{э.п}=W'_{c}+\frac{f_{c}p_{c}}{\dot{m}}. \quad (1.4)$$

One should emphasize that thrust in the vacuum - this purely reaction

force; it is actual thrust characteristics of RD, since it is wholly determined by the processes, which occur intra-chamber/intra-camera and leading to an increase in the momentum of working medium/propellant. The thrust of thermal rocket engines in the vacuum is expressed also by the equation

$$P_п = K_P p_к f_{кр}, \qquad (1.5)$$

where K_P - thrust coefficient in the vacuum, which shows, in how often the thrust in the vacuum of more than product $p_к f_{кр}$, i.e.,

$$K_P = \frac{P_п}{p_к f_{кр}}, \qquad (1.6)$$

where $p_к$ - gas pressure in the combustion chamber (it is more accurate, at the nozzle entry);

$f_{кр}$ - area of the critical (smallest) cross section of nozzle.

Thrust coefficient in the vacuum - dimensionless quantity, which depends on the characteristics of nozzle. Value K_P increases/grows with an increase of the ratio of pressure in the cross section at the nozzle entry and in nozzle exit section, and also with the decrease of losses in the nozzle. Usually $K_P = 1,2 \div 2,2$.

Rocket engine thrust when ambient pressure is present,. Let us determine value and direction of the force, which affects on the engine from the side of the environment.

chamber/camera is equal and it is equal to the pressure of the undisturbed environment. For example, for the engine, which works in the atmosphere of the Earth, the pressure indicated is uniquely determined by height/altitude h above its surface, and for the engine, which works in the water, by submersion depth of apparatus.

Pressure at height/altitude h let us designate p_h and the thrust which develops FD at height/altitude h, through P_h.

Let us incidentally note that during the motion of rocket apparatus in the atmosphere of the Earth or another planet on the housing of apparatus acts the aerodynamic drag, which brakes the motion of apparatus.

Let us determine resultant force of pressure on the external surface of chamber walls.

The resultant force of the evenly distributed ambient pressure on any closed vessel is equal to zero.

Page 13.

But chamber/camera is vessel with the opening/aperture, namely with open exit section with an area of f_c. Therefore the equilibrium of forces of ambient pressure is disturbed: appears the force, equal to the product of area f_c to the ambient pressure p_h. Let us designate the force indicated, caused by the presence of the barometric pressure of the atmosphere, $P_{бар\,h}$.

Consequently,

$$P_{бар\,h} = p_h f_c. \qquad (1.7)$$

As it is not difficult to see from Fig. 1.8, force $P_{бар\,h}$ is directed to the side, opposite to the sense of the vector of reaction force. Therefore taking into account equality (1.7) thrust at the arbitrary height/altitude h

$$P_h = P_п - p_h f_c \qquad (1.8)$$

or taking into account equation (1.2)

$$P_h = \dot{m}W_c + f_c(p_c - p_h). \qquad (1.9)$$

Consequently, rocket engine thrust in any mode of its operation which is characterized by constant flow rate/consumption of the working medium/propellant $\dot{m}$, it takes the different values with a change in pressure p_h, i.e. flight altitude of vehicle.

Equation (1.9) expresses the thrust of any rocket engine with its work at the arbitrary height/altitude. The thrust indicated does not depend on the flight speed of vehicle and with an increase in

altitude, i.e., with the decrease of pressure p_h to a certain extent increases/grows, what is the advantage of rocket engines in comparison with the jet engines, which use as the working medium/propellant the environment.

Effective discharge velocity at the arbitrary height/altitude h is equal to

$$W_{\text{э}h}=\frac{P_h}{\dot{m}} \qquad (1.10)$$

or taking into account (1.3) and (1.8)

$$W_{\text{э}h}=W_{\text{э.п}}-\frac{p_h f_c}{\dot{m}}. \qquad (1.11)$$

It was above indicated that important rocket-motor characteristics is the thrust in vacuum $P_{\text{п}}$. For the engines of the flight vehicles which are started from the Earth either from the surface ships, high value has also thrust at the level of sea, or thrust in Earth $P_{\text{з}}$.

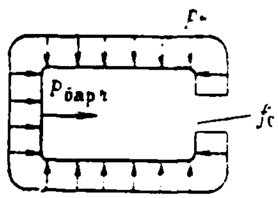


Fig. 1.8. Diagrams/curves of the forces of pressure of the environment (atmosphere) on the external surface of the walls of cylindrical container with the opening/aperture with an area of f_c in one bottom.

Page 14.

If we designate the pressure of the atmosphere at the level of sea p_3, then in accordance with equation (1.8) thrust at the level of sea

$$P_3 = P_п - p_3 f_c \qquad (1.12)$$

or taking into account equation (1.2)

$$P_3 = \dot{m} W_c + f_c (p_c - p_3), \qquad (1.13)$$

i.e., due to the presence of the pressure of the atmosphere thrust at the level of sea is not the actual characteristic strictly of engine.

Rocket engines are characterized not only by thrust, but also by power.

Most important power rocket-motor characteristics is the power of exhaust jet. This is net power, directly utilized for the thrust application. For the case of work in the vacuum the power of exhaust jet

$$N_{\text{стр. п}} = \frac{\dot{m}W_{\text{а. п}}^{2}}{2}. \qquad (1.14)$$

In any engine net power composes certain part of the source power of primary energy $N_{\text{перв}}$, determined by the efficiency of engine η. Therefore for any rocket engine

$$N_{\text{перв}} = \frac{N_{\text{стр. п}}}{\eta}.$$

In the chemical rocket engines (KhRD) the source of primary energy is the fuel/propellant. Therefore their primary power is determined by the consumption of fuel $\dot{m}$ and by chemical energy $E_{\text{хим}}$, of that containing in 1 kg. of fuel/propellant, i.e.,

$$N_{\text{перв}} = \dot{m}E_{\text{хим}}.$$

For the work of all nonchemical (nuclear, solar and electrical) rocket engines on board the vehicle necessary is the source of primary energy. With the increase of the necessary power coefficient of exhaust jet of such engines respectively (taking into account the efficiency of engine η) it is necessary to increase the source power of primary energy. Therefore its mass and dimensions for the creation of the large power of exhaust jet of all types nonchemical RD, especially at low values η, become exaggerated, which makes the

characteristics worse of rocket vehicle as a whole and to a considerable degree limits the range of the utilized thrusts of engines indicated.

§ 1.4. Specific parameters of rocket engines.

The most important parameters of RD include: a) specific impulse; b) the specific expenditure/consumption of working medium/propellant; c) the specific power of exhaust jet and d) the specific mass of engine.

Page 15.

Specific impulse. Is distinguished mass and density specific impulse.

Mass specific impulse I_{ym} is called the impulse/momentum/pulse, per unit of the mass (1 kg.) of working medium/propellant,

$$I_{y\text{м}} = \frac{I}{m}. \qquad (1.15)$$

If the thrust of engine P is constant during entire operating time of engine τ, then the impulse/momentum/pulse, developed with engine, is equal to

$$I = P\tau. \qquad (1.16)$$

Substituting equality (1.15) in equation (1.16), we obtain

$$I_{y\text{м}} = \frac{P\tau}{m} = \frac{P^{*}}{m}, \qquad (1.17)$$

where $\dot{m}$ - mass flow rate per second of working medium/propellant.

FOOTNOTE [1]. Equation (1.17) is conveniently used in the theoretical analyses; it partly explains the utilization of a term "specific thrust" (thrust, which falls to the expenditure/consumption of working medium/propellant, equal to 1 kg/s). ENDFOOTNOTE.

If with the work of engine its thrust changes, then can change also specific impulse; in this case is used the concept "average/mean specific impulse" $I_{уд.ср}$. If for the time of testing τ the engine thrust changes according to the law of $P=f(\tau)$ (Fig. 1.9), then the value of area under curve $P=f(\tau)$, numerical equal to the value of integral $\int_0^\tau P d\tau$, is total jet firing; let us designate it I_Σ. Then

$$I_{уд.ср}=\frac{I_\Sigma}{\Delta m},$$

where Δm - total quantity of working medium/propellant, spent by engine for the operating time.

As can be seen from Fig. 1.9, value I_Σ can be written in the form of the equation

$$I_\Sigma=P_{ср}\tau,$$

where $P_{ср}$ - mean-integral thrust value for production time of engine.

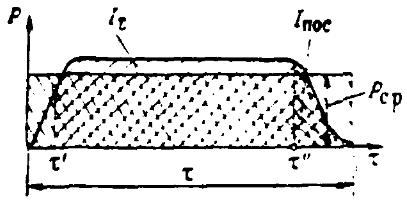


Fig. 1.9. The graph of a change in the thrust of RD in time P=f(τ): τ - total operating time of RD; τ' - moment of separation of rocket from the launcher; τ'' - moment of supplying command/crew to the disconnection of RD.

Page 16.

In accordance with equation (1.17) the specific impulse of RD depends on ambient pressure, since it influences thrust. Therefore in the most general case specific impulse it is necessary to designate $I_{уд\,h}$ by underscoring thereby its dependence on height/altitude h, so that equation (1.17) acquires the following form:

$$I_{уд\,h}=\frac{P_h}{m} \qquad (1.18)$$

or taking into account equation (1.9)

$$I_{уд\,h}=W_c+\frac{f_c(p_c-p_h)}{m}. \qquad (1.19)$$

On the basis of equations (1.10) and (1.18)

$$I_{уд\,h}=W_{э\,h}. \qquad (1.20)$$

The greatest specific impulse of RD is developed with its work

in the vacuum. Specific impulse in vacuum $I_{уд.п}$ is the most important parameter of RD, characterizing the effectiveness of the applied working medium/propellant and the perfection of the construction/design of engine.

Taking into account equations (1.18), (1.19) and (1.20) specific impulse in the vacuum and at the level of sea

$$I_{уд.п}=\frac{P_п}{\dot{m}}; \qquad (1.21)$$

$$I_{уд.п}=W_c+\frac{p_c f_c}{\dot{m}}=W_{э.п}; \qquad (1.22)$$

$$I_{уд.з}=\frac{P_з}{\dot{m}}; \qquad (1.23)$$

$$I_{уд.з}=W_c+\frac{f_c(p_c-p_з)}{\dot{m}}=W_{э.з}. \qquad (1.24)$$

Although the value of specific impulse at the level of sea is not the actual characteristic strictly of engine, it they use extensively, which is explained by the following reasons.

1. Many engines operate (work out) under terrestrial conditions, i.e., at pressure $p_з$.

2. Specific impulse at the level of sea can be designed according to results of bench test of engine on level (1.23). Thrust level $P_з$ and the mass flow rate of the working medium/propellant $\dot{m}$ can be measured in the process of bench test of engine with the sufficiently low error.

3. Conducting tests of engines on earth/ground when $p_н \to 0$, i.e., with safeguard of vacuum around engine chamber, is very hindered/hampered.

Page 17.

Therefore specific impulse in the vacuum is determined via conversion, using as the initial value a specific impulse at the level of sea, designed according to the results of bench test of engine.

The dimension of specific impulse can be determined from equation (1.18):

$$[I_{уд}] = \frac{[P]}{[\dot{m}]} = \frac{[н]}{^{(1)}[кг/сек]} = \left[\frac{н\cdot сек^{(2)}}{кг^{(3)}}\right].$$

Key: (1). kg/s. (2). N•s. (3). kg.

In the system MKGSS the force and mass flow rate have a dimension kgf and kg/s [1].

FOOTNOTE [1]. Here and subsequently as the unit of mass in the old system of unity we select kilogram (kg), but not kg•s²/m (the derived unit of mass in the system MKGSS). ENDFOOTNOTE.

Therefore the dimension of specific impulse in this system

$$I_{y\text{д}}=\frac{[P]}{[\dot m]}=\frac{(1)[\kappa\Gamma]}{(2)[\kappa z\ ce\kappa]}=\overset{(3)}{\left[\frac{\kappa\Gamma\cdot ce\kappa}{(4)\ \kappa z}\right]}.$$

Key: (1). kgf. (2). kg/s. (3). kg•s. (4). kg.

However, in the technical literature frequently is used another dimension - s, which is valid only for the weight specific jet firing, which works at the level of sea,

$$I_{y\text{д.з}}=\frac{P_3}{G},$$

i.e. for the thrust, per unit of weight flow rate per second of the working medium/propellant G,

$$[I_{y\text{д.з}}]=\frac{[P_3]}{[G]}=\frac{(1)[\kappa\Gamma]}{(2)[\kappa\Gamma/ce\kappa]}\overset{(3)}{=}[ce\kappa].$$

Key: (1). kgf. (2). kgf/s. (3). s.

The expression of specific impulse at the arbitrary height/altitude and those in the vacuum in the seconds is artificial, conventional method, since in this case, for example, the thrust in the vacuum, expressed in the kilograms of force (kgf), is carried to the flow rate per second of working medium/propellant, expressed in the units of weight on the Earth.

The value of specific impulse substantially affects the expenditure/consumption of the working medium/propellant $\dot m$, necessary

for the creation of the prescribed/assigned thrust. From equation (1.18) it follows that with the increase of specific impulse value $\dot{m}$ for the creation of one and the same thrust is decreased.

Density specific impulse is called the impulse/momentum/pulse, which falls per unit volume (1 m^3) of the working medium/propellant V,

$$I_{уд.об} = \frac{I}{V}$$

or it is analogous with equation (1.18)

$$I_{уд.об\,h} = \frac{P_h}{\dot{v}}, \qquad (1.25)$$

where $\dot{v}$ - volumetric flow rate per second of working medium/propellant in the m^3/s.

Page 18.

On the basis of equation (1.25) dimension $I_{уд.об}$ in the system of SI

$$[I_{уд.об\,h}] = \frac{[P_h]}{[\dot{v}]} = \frac{[н]}{^{(1)}[м^3/сек]} \overset{(2)}{=} \left[\frac{н \cdot сек}{м^3}\right]$$

Key: (1). m^3/s. (2). N•s.

and in the system MKGSS [1]

$$[I_{уд.об\,h}] = \frac{[P_h]}{[\dot{v}]} = \frac{^{(1)}[кГ]}{^{(2)}[л/сек]} \overset{(3)}{=} \left[\frac{кГ \cdot сек}{л}\right].$$

Key: (1). kgf. (2). l/s. (3). kg•s.

FOOTNOTE [1]. Here and subsequently as the volume unit in the dimension of specific volume impulse in the old system of unity we select liter (l), but not m^3 (volume unit in the system MKGSS). ENDFOOTNOTE.

Density specific impulse is connected with the mass specific impulse with the following relationship/ratio:

$$I_{\text{уд.об}h}=I_{\text{уд}h}\varrho_{\text{т}}, \qquad (1.26)$$

where $\varrho_{\text{т}}$ - density of working medium/propellant (fuel/propellant) in kg/m^3.

Specific expenditure/consumption of working medium/propellant. The specific expenditure/consumption of working medium/propellant is called the quantity of working medium/propellant, expended by rocket engine in 1 s on 1 n [1 kgf] of the developed with it thrust,

$$C_{\text{уд}h}=\frac{\dot{m}}{P_h}. \qquad (1.27)$$

As can be seen from the comparison of equations (1.18) and (1.27), parameters $I_{\text{уд}h}$ and $C_{\text{уд}h}$ are inversely proportional values. Therefore $C_{\text{уд}h}$ has in the system of SI dimension kg/(N•s), and in the system MKGSS - kg/(kg•s).

The specific expenditure of rocket engines is considerably more than in the engines, which use as the working medium/propellant to only the substance, placed on board the vehicle, but also the environment.

The specific expenditure of the working medium/propellant of jet engine (VRD) is equal to the ratio of the fuel consumption $\dot{m}_{r}$ per second to thrust P_{h}. The air flow rate through VRD many times exceeds fuel consumption, however, since air is taken from the atmosphere, it in the specific expenditure VRD they do not include.

Into the specific expenditure of rocket engines enters entire mass of the rejected working medium/propellant, since increasingly working body is located on the vehicle itself.

Due to the high specific expenditure the operating time of many rocket engines is comparatively small: it does not usually exceed several hundred seconds.

The specific power of exhaust jet. The specific power of exhaust jet is called the power of the stream of working medium/propellant, per unit of the reaction force of engine, i.e., for the case of the work of engine in the vacuum

$$N_{\text{стр.уд.п}}=\frac{N_{\text{стр.п}}}{P_{\text{п}}}. \quad (1.28)$$

Page 19.

The specific power of exhaust jet as any specific power, has a dimension of W/N in the system of SI and kg•m/(s•kg) in the system MKGSS, and in the reduced form in both systems - m/s.

Substituting expressions (1.3) and (1.14) in equation (1.28), we obtain

$$N_{\text{стр.уд.п}}=\frac{\dot{m}W^2_{э.п}}{2\dot{m}W_{э.п}}=\frac{W_{э.п}}{2}. \qquad (1.29)$$

From equations (1.29) and (1.20) with the entire obviousness it follows that with an increase in values $W_{э.п}$ and $I_{уд.п}$ increases/grows the power of exhaust jet, which falls on 1 N [1 kgf] of reaction force, i.e., increase/grow the expenditures of power for the work of the engine of the prescribed/assigned thrust. Value $N_{\text{стр.уд.п}}$ characterizes a difference in all types of RD in the relation to the expenditures of power and together with values and $I_{уд.п}$ - the effectiveness of the utilization of engines in the flight vehicle.

Specific mass of RD. Specific mass of RD is called the mass of engine during its work, per unit the thrust, developed with it in the vacuum, i.e., if we designate specific mass of RD $g_{дв}$, then

$$g_{дв}=\frac{m_{дв}}{P_п}, \qquad (1.30)$$

where $m_{дв}$ - mass of engine during its work, which encompasses not only dry mass, but also the mass of working medium/propellant, which takes place with the work on mains and assemblies of engine.

In proportion to the construction-engineering perfection of each type of engine its specific mass descends, which makes it possible to increase the flying range of rocket with the same mass of payload or to increase the mass of the payload of rocket with the same distance of its flight. The approximate values of the specific mass of different types of RD are shown in Table 1.1 (see § 1.6).

§ 1.5. Other parameters of rocket engines.

Besides the thrust and the specific parameters, rocket engine characterize following data.

1. Type of working medium/propellant. Each RD they design for the completely specific working medium/propellant, utilized for creation of thrust, moreover from it to a considerable degree depend the specific parameters of engine and the effectiveness of its use/application in the rocket vehicle.

2. Operating time of engine τ. Usually for ZhRD it does not exceed 1000 s, but for RDTT - 150 s. The engines of some types of rocket vehicles must be included repeated or multiple. For such engines are given not only the time of continuous operation upon each

inclusion/connection and the number of inclusions/connections, but also the required or permissible time interval between the inclusions/connections.

Page 20.

3. Thrust range. For the accomplishment of the objective of rocket vehicle frequently appears the need in a change in the thrust level of its operating engine relative to rated thrust. Nominal is called the thrust which develops the engine with that expenditure of working medium/propellant, for whom is designed the engine. It should be noted that rated thrust of RD in the Earth and in the vacuum has different values. By changing the expenditure of working medium/propellant it is possible to respectively change the engine thrust with its work both on one and the same and at any height/altitude. The thrust range of RD is been given in the percentages of rated thrust (for example, 10-100o/o) or by the relation (for example, 10:1), which shows, in how often the thrust level of engine can descend in comparison with its nominal value.

4. Pressure of gaseous working medium/propellant or gaseous reaction products propellant components at nozzle entry and at output/yield from it ($p_{к}$ and p_{c} into bar [kgf/cm²]).

Value p_c and nozzle exit section is preliminarily examined in § 1.3. Concept "nozzle exit section" for the different types of rocket engines has its value. For the chambers/cameras thermal RD nozzle exit section is the nozzle edge, i.e., the final cross section of its solid walls, after which gas flow flows out into the environment. In this cross section is finished the mechanical reaction of gas flow and nozzle liners. Nozzle exit section electrical RD is the cross section (or surface), after which ceases the reaction between electrostatic or electromagnetic field of motor and flow of the working medium/propellant, which escapes from the motor.

Chamber of thermal RD can be conditionally divided into two parts. In its initial part to the working medium/propellant is supplied the heat, while in chemical RD occurs the chemical reaction of burning or decomposition of fuel/propellant. This part of the chamber/camera is the chamber/camera of heating, and in connection with chemical RD - by combustion chamber or decomposition. Here in essence is finished the process of heating working medium/propellant, and for chemical RD - process of burning or decomposition of fuel/propellant.

In the final part of the chamber/camera, which, as has already been indicated, is called nozzle, occurs the expansion of the gaseous products of heating working medium/propellant or chemical reaction

product. Cross section at the nozzle entry is the cross section, which divides these two parts.

Chamber of electrical RD also can be conditionally divided into two parts. In the initial part working the body is prepared for the subsequent dispersal/acceleration, for which it usually vaporizes, is heated and is ionized. The final part of the chamber/camera is the motor, in which is accomplished/realized the dispersal/acceleration of electrically charged particles of the working medium/propellant.

Page 21.

The parameters of the products of heating working medium/propellant or chemical reaction product at the nozzle entry subsequently we will designate by index k "" ($p_к$, $T_к$, $\varrho_к$, $W_к$ and so forth) in contrast to the parameters at the nozzle outlet which are designated, as has already been indicated, by index s "" (p_c, T_c, ϱ_c, W_c and so forth).

5. Total impulse of thrust I_Σ in N•s [kg•s] or in kN•s [T•s]. As has already been indicated, the value of the total impulse of thrust is numerically equal to area under graph P=f (τ). In zero time ($\tau=\tau'$) it is necessary to select the moment/torque of the start of the vehicle (see Fig. 1.9).

6. Impulse/momentum/pulse of consequence $I_{пос}$ in N•s [kg•s] or in kN•s[T•s]. The impulse/momentum/pulse of consequence is called the impulse/momentum/pulse, developed with engine after the delivery of the command to its inclusion/connection τ'' (see Fig. 1.9). Usually during the design of RD they attempt to decrease value $I_{пос}$ and especially its spread, since in this case is decreased the scatter of the velocity of vehicle after engine cutoff, which facilitates accomplishing mission objective.

Thermal type nuclear, solar and electrical RD additionally characterize by the source power (receiver) of primary energy, which is converted into the heat and is used for heating of working medium/propellant, and YaRD - also by type of nuclear fuel/propellant (for example, fissionable material).

§ 1.6. Classification of rocket engines.

By the final goal of all processes, which take place in the rocket engines, is the creation of the greatest kinetic jet energy by accelerating the working medium/propellant in some manner or another.

Rocket engines classify according to the type of their primary

energy to the chemical, nuclear, solar and electrical, and also steam-gas (Fig. 1.10).

Chemical energy possess, as has already been indicated, the substances which can enter into the chemical reactions, which take place with the liberation of heat and the formation of gaseous products. As a result of chemical reactions can be formed also the products in the liquid or solid state, what is usually undesirable.

The examples of such chemical reactions are the reaction of the reaction of oxygen (oxidizer) and hydrogen (fuel) - the reaction of oxidation or burning

$$\frac{1}{2}O_2 + H_2 \rightarrow H_2O$$

and the decomposition reaction of peroxide of hydrogen

$$H_2O_2 \rightarrow H_2O + \frac{1}{2}O_2.$$

Page 22.

Chemical rocket engines are the only type of familiar RD whose working medium/propellants (propellant components) are simultaneously the source of heat and mass of the rejected substance, which in this case are the reaction products of the reaction of propellant components.

The initial components of chemical fuel/propellant can be liquid, solid, gaseous, gel-like and fluidized [23]. In chemical RD can be used one, two are considerably less frequent than three propellant components, their initial state of aggregation can be one and the same or different.

The rocket engines of hybrid fuel/propellant (RDGT) work on the propellant components, which have different initial state of aggregation. In RDGT it is possible to use a solid-liquid, liquid-gas and solid-gas fuel/propellant. However, in essence at present are developed DU with RDGT on the hybrid propellant (Fig. 1.11).

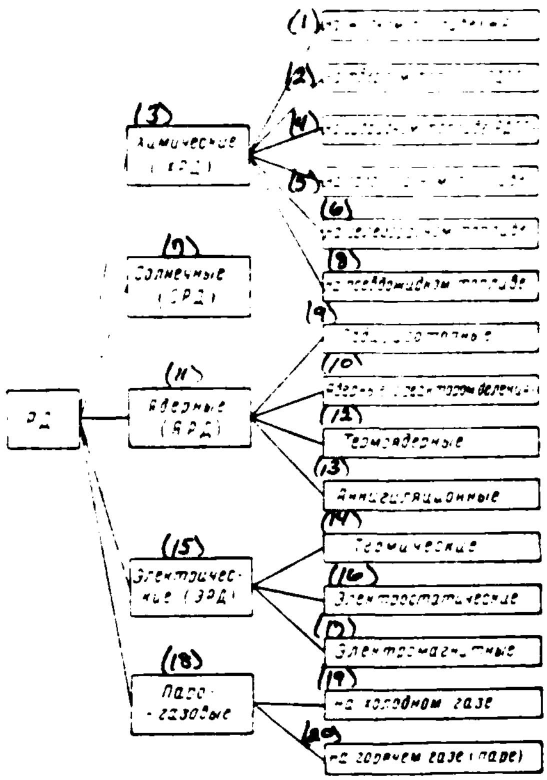


Fig. 1.10. General/common/total classification of rocket engines.

Key: (1). On the liquid propellant. (2). On solid fuel. (3). Chemical. (4). On hybrid fuel/propellant. (5). On gaseous fuel. (6). On gel-like fuel/propellant. (7). Solar. (8). On fluidized fuel/propellant. (9). Radioisotope. (10). Nuclear (with reactor of division). (11). Nuclear. (12). Thermonuclear. (13). Annihilation. (14). Thermal. (15). Electrical. (16). Electrostatic. (17). Electromagnetic. (18). Steam-gas. (19). On cold gas. (20). On hot gas

Page 23.

In the chamber/camera of such RDGT place the charge of solid component the propellants (is usually it fuel); another (liquid) component is supplied into the chamber/camera from the tank with the aid of the special feed system.

The steam-gas RD (Fig. 1.12) work on the cold or hot high-pressure (vapor) gas, previously stored up in the special tank/balloon or the chamber/camera. In these engines primary energy is the energy of thermal agitation and potential energy of compressed gas. For the thrust application in gas RD it is necessary to only drive away gas, i.e., in them the primary energy indicated is converted directly (without the intermediate forms of energy as in all other thermal, and also electrical RD) into the kinetic jet energy. Therefore gas RD have simple construction/design.

For the work of SRD (solar rocket engines), ERD and YaRD on board the rocket vehicle necessary to have not only stored up working medium/propellant, but also the source of primary energy or the concentrator of external energy.

Solar energy - is energy of the electromagnetic radiation of the sun; it it is possible to use for heating of working medium/propellant, for example, via the focusing of solar rays/beams on any absorber of heat through which it flows/occurs/lasts.

Into composition of DU with solar RD (Fig. 1.13) enter tank with the working medium/propellant, the absorber of heat - heat exchanger, reflector for the focusing of solar rays/beams on the absorber and chamber/camera.

In thermal type ERD electrical energy it is converted in the special devices into the heat. Such devices they can be:

1) the ohmic resistance (resistor) which with the course of electric current is heated to the high temperature (Fig. 1.14);

2) the electric arc, which also isolates during its excitation heat (Fig. 1.15).

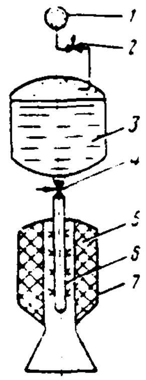


Fig. 1.11.

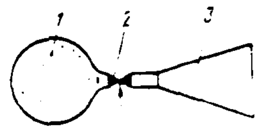


Fig. 1.12.

Fig. 1.11. Diagram of DU with RDGT: 1 - tank/balloon with compressed gas; 2, 4 - valves; 3 - tank with liquid oxidizer; 5 - solid-propellant grain; 6 - injecting device; 7 - housing.

Fig. 1.12. Diagram gas RD: 1 - tank/balloon with compressed gas; 2 - valve; 3 - nozzle.

Page 24.

With respect to the devices or ERD of thermal type indicated they subdivide into the resistor ones and the electric arc ones.

Working body is heated by the heat, isolated by resistor or electric arc, to one or the other temperature, after which heating products are accelerated/dispersed in the nozzle and flow out from it, creating thrust.

Is possible the use/application of thermal type of ERD with the exploding wires (see pg. 44).

In the electrical rocket engines the electrical energy is used for the creation of electrically charged particles (ions, free electrons) and for their dispersal/acceleration with the aid of electrostatic or electromagnetic field. With respect to this RD indicated subdivide into the electrostatic ones and the electromagnetic ones. Working body of ERD (with exception of thermal ones) must possess in the motor the electrical properties.

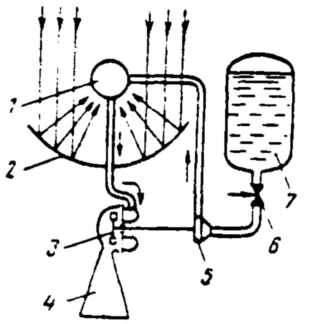


Fig. 1.13.

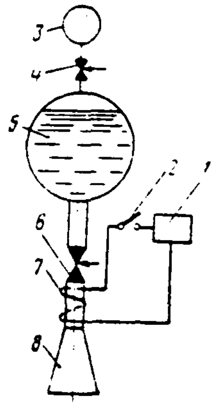


Fig. 1.14.

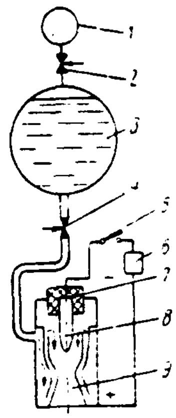


Fig. 1.15.

Fig. 1.13. Diagram of DU with SRD: 1 - absorber of heat - heat exchanger; 2 - reflector; 3 - turbine; 4 - nozzle; 5 - centrifugal pump; 6 - valve; 7 - tank with liquid working medium/propellant.

Fig. 1.14. Diagram of DU with resistor RD: 1 - electric power source; 2 - relay-switch; 3 - tank/balloon with compressed gas; 4, 6 - valves; 5 - tank with liquid working medium/propellant; 7 - resistor (electric heater in the form of ohmic resistance); 8 - nozzle.

Fig. 1.15. Diagram of DU with electric arc RD: 1 - tank/balloon with compressed gas; 2, 4 - valves; 3 - tank with liquid working medium/propellant; 5 - relay-switch; 6 - electric power source; 7 - insulator; 8 - cathode; 9 - anode-nozzle.

Page 25.

In the method of dispersing/accelerating the working medium/propellant of ERD in principle they differ from KhRD, YaRD, SRD and gas RD, in which working the body is accelerated as a result of the energy conversion of thermal agitation and potential energy of compressed gas into the kinetic jet energy; in particular, the high values of specific impulse of ERD[1] can be ensured without heating of working medium/propellant to the high temperature.

FOOTNOTE 1. Subsequently by the electrical rocket engine (ERD), if are absent special stipulations, are understood all types of ERD, except thermal ones. Thermal type of ERD call also the electrothermal rocket engines (ETRD) [23]. ENDFOOTNOTE.

The effectiveness of apparatus with any type of ERD to a considerable degree depends on the characteristics of the source of electrical energy. It must possess large specific output power, i.e., with the large electrical power, produced by source per unit of mass and volume.

The major advantages of ERD are high discharge velocity W_e and, consequently, large specific impulse. But ERD just like SRD, it is inexpedient to create for obtaining the high thrusts due to necessary

in this case power increase of the source of electrical energy (or the receiver of external energy) and, consequently, also its mass up to the values, not admitted from the point of view of the characteristics of rocket apparatus as a whole.

Necessary specific primary power of DU with ERD is determined according to the equation

$$N_{\text{перв.уд}}=\frac{W_{\text{э.п}}}{2\eta_{\text{уск}}\eta_{\text{эу}}},$$

where $\eta_{\text{уск}}$ — efficiency of the accelerator (motor) of ERD; $\eta_{\text{эу}}$ — efficiency of the source of electrical energy (power unit).

If the efficiency of accelerator has sufficiently high values (approximately/exemplarily 0.5-0.6), then due to the special working conditions in outer space the efficiency of the source of electrical energy it is usually low (approximately/exemplarily 0.1-0.2). High values $W_{\text{э.п}}$ and low values $\eta_{\text{уск}}$ give rise to the large necessary source power of electrical energy and source of primary energy of rocket apparatus with ERD.

As a result of the large specific impulse and the low thrust the expenditure/consumption of working medium/propellant of ERD is very low. Therefore the time of continuous operation of ERD can be very large (of up to several years).

In the nuclear rocket engines can be used the energy of radioactive decay, division and nuclear fusion of special substances, and also annihilation.

Decomposition/decay of radioisotopes, which takes place spontaneously, is the source of heat. RD, in which is used the energy of radioactive decay, are called radioisotope.

Page 26.

The nuclear fission of fissionable materials occurs under conditions, artificially created in the fission-type nuclear reactors, in which are used the substances with the large atomic mass. RD with the fission-type nuclear reactor frequently call simply nuclear.

Nuclear mass during the radioactive decay and the division is decreased ("mass defect").

The synthesis (merging/coalescence) of the nuclei of substances with the low atomic mass can occur under the effect of the very high temperatures. The fusion reaction and the engines, in which it flows/occurs/lasts, call thermonuclear. The example to thermonuclear fusion is the mucous membrane of heavy hydrogen (deuterium) with the

formation of helium. Working medium/propellant th thermonuclear RD is gas with the very high temperature (to $5 \cdot 10^{8}$ °K), since only at this temperature kinetic energy of the converging nuclei with the low mass can exceed energy of the forces of their mutual repulsion. Such high-temperature extinguished (it they call plasma) contains the electrically charged/loaded parts of the molecules and atoms, but in each volume element it electrically neutral.

The special feature/peculiarity of thermonuclear fusion lies in the fact that the mass of the product nuclei of helium to the noticeable value is lower than the mass of the initial nuclei of hydrogen. The "mass defect indicated" in the presence of the thermonuclear fusions is significantly more than during the radioactive decay or the nuclear fission. With respect to this as a result of thermonuclear fusion is isolated a considerably greater quantity of heat. For the safeguard of plausible working conditions of engine block high-temperature plasma must be held far from its walls with the aid of the magnetic field.

In contrast to fission reaction of nuclei the thermonuclear fusion does not lead to the radioactive contamination of the engine components and rocket apparatus, or environment.

Thermonuclear fusions occur on the sun and other stars. Under

artificial conditions the thermonuclear fusion is accomplished/realized at present only in the form of explosion in the thermonuclear (hydrogen) bombs. Control capability of thermonuclear fusion is the very first task of contemporary applied nuclear physics.

The reaction of annihilation appears during the supplying into the chamber/camera annihilator) particle of substance (for example, electrons) and particles of the antimatter (for example, positrons). With the connection of the particles indicated the energy, which is contained in their mass, completely is converted into the radiant energy on the relationship/ratio

$$E=mc^2,$$

where E - radiant energy; m - initial rest mass; c - speed of light in the vacuum, equal to approximately/exemplarily $3\cdot10^8$ m/s.

Page 27.

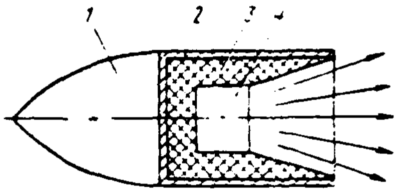
YaRD, in which the heat, isolated in the presence of the nuclear reactions, is used for heating of working medium/propellant, is called nuclear engines with the heat transfer to the working medium/propellant, or heat exchange YaRD. Exhaust jet of YaRD can be the mixture of the products of the reacted nuclear fuel/propellant and working medium/propellant. Such YaRD are called mixture.

Is not excluded the possibility of designing of the nuclear engines whose exhaust jet consists of the nuclear particles, which are obtained during the radioactive decay, division or nuclear fusion [21]. For example, are known the projects of radioisotope sail - the rocket engine which creates thrust as a result of the nuclear decomposition of the radioisotope, which covers material of sail (from the plastic or the metal), and the formation of the anisotropic flow of α-particles (atomic nuclei of helium).

Photon, or quantum, are called YaRD whose exhaust jet is not the flow of molecules, ions, atoms or their particles, but the flow of photons (electromagnetic quanta, or light quanta). In the photon engines the rest energy (mass) of nuclear fuel/propellant completely is converted into the radiant energy, i.e., photon engines create thrust as a result of the radiation/emission.

Diagrams of DU with thermonuclear photon and annihilation of RD are depicted in Fig. 1.16 and 1.17.

Photon engines possess theoretically maximum characteristics among all types of RD, since exhaust jet, which is light quanta, possesses the maximally attainable speed (speed of light in the vacuum).



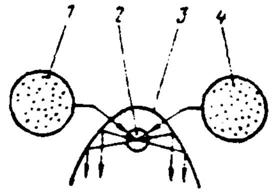


Fig. 1.16.

Fig. 1.17.

Fig. 1.16. Schematic of thermonuclear photon engine: 1 - section with payload; 2 - housing; 3 - shaped absorber of photons; 4 - thermonuclear source of photons.

Fig. 1.17. Diagram of DU with annihilation engine: 1 - tank with substance; 2 - annihilator; 3 - parabolic reflector; 4 - tank with antimatter.

Page 23.

Table 1.1. Comparative table of the characteristics of different types of RD.

(1) Тип РД	$I_{уд.п}$ [2] (4) $\frac{н \cdot сек}{кг}$ (5)	$I_{уд.п}$ [2] (6) $\frac{кГ \cdot сек}{кг}$ (5)	$N_{стр.уд.п}$ (7) $\frac{вт}{н}$ (8)	$N_{стр.уд.п}$ (7) $\frac{вт}{кГ}$ (9)	(2) Удельная масса РД [2] (5) $\frac{кг}{н}$ (8)	(2) Удельная масса РД [2] (5) $\frac{кг}{кГ}$ (9)	(3) Особенности РД, определяющие эффективность их применения на ракетном аппарате	
ЖРД	До $5 \cdot 10^3$	До $5 \cdot 10^2$	До $2,5 \cdot 10^3$	До $2,5 \cdot 10^4$	0,001	0,01	Возможность изменения тяги в широком диапазоне (10)	(13) Возможность создания больших тяг
РДТТ	До $3 \cdot 10^3$	До $3 \cdot 10^2$	До $1,5 \cdot 10^3$	До $1,5 \cdot 10^4$			Простота конструкции (11)	
РДГТ	До $5 \cdot 10^3$	До $5 \cdot 10^2$	До $2,5 \cdot 10^3$	До $2,5 \cdot 10^4$			Возможность изменения тяги (12)	
Газовые (14)	До 10^3	До 10^2	До $0,5 \cdot 10^3$	До $0,5 \cdot 10^4$	—	—	1. Чрезвычайная простота конструкции (15) 2. Весьма высокая надежность (16)	(22) Возможность использования рабочего тела с малым значением μ
ЯРД с твердофазным реактором (17)	От $8,0 \cdot 10^3$ до $1,2 \cdot 10^4$	От $8,0 \cdot 10^2$ до $1,2 \cdot 10^3$	От $4,0 \cdot 10^3$ до $0,6 \cdot 10^4$	От $4,0 \cdot 10^4$ до $0,6 \cdot 10^5$	0,01	0,1	1. Высокие значения $I_{уд}$ (19) 2. Возможность создания больших тяг (20)	
ЭРД термического типа (18)	До $2,5 \cdot 10^4$	До $2,5 \cdot 10^3$	До $1,25 \cdot 10^4$	До $1,25 \cdot 10^5$	От $2,5 \cdot 10$ до $2,5 \cdot 10^2$	От $2,5 \cdot 10^2$ до $2,5 \cdot 10^3$	1. Высокие значения $I_{уд}$ (19) 2. Возможность длительной работы (21)	
СРД	От $8,0 \cdot 10^3$ до $1,2 \cdot 10^4$	От $8,0 \cdot 10^2$ до $1,2 \cdot 10^3$	От $4,0 \cdot 10^3$ до $0,6 \cdot 10^4$	От $4,0 \cdot 10^4$ до $0,6 \cdot 10^5$	10	10^2		
Электростатические РД (23)	От $5,0 \cdot 10^4$ до 10^6	От $5,0 \cdot 10^3$ до 10^5	От $2,5 \cdot 10^4$ до $5 \cdot 10^5$	От $2,5 \cdot 10^5$ до $5,0 \cdot 10^6$	От $5 \cdot 10$ до 10^4	От $5 \cdot 10^2$ до 10^5	1. Высокие значения $I_{уд}$ (19) 2. Возможность длительной (до нескольких лет) работы (25) 3. Отсутствие необходимости в нагреве рабочего тела до высокой температуры (26)	
Электромагнитные РД (24)	От $5,0 \cdot 10^4$ до 10^5	От $5,0 \cdot 10^3$ до 10^4	От $2,5 \cdot 10^4$ до $5,0 \cdot 10^4$	От $2,5 \cdot 10^5$ до $5,0 \cdot 10^5$	От $5 \cdot 10$ до 10^3	От $5 \cdot 10^2$ до 10^4		

Page 29.

1 Тип РД	Особенности РД, ограничивающие их применение в ракетных аппаратах (27)	Степень освоения (28)
ЖРД	(29) Сравнительно малое значение $I_{уд}$ из-за малого количества теплоты, выделяющейся при химических реакциях, и большой величины μ продуктов химических реакций	Освоены и широко применяются (30)
РДТТ		
РДГТ		Осваиваются; применяются редко (31)
14 Газовые	1. Малое значение $I_{уд}$ (32) 2. Большая удельная масса (33) 3. Малая тяга (34)	Освоены; применяются в искусственных спутниках Земли и в космических аппаратах для создания малых тяг (35)
17 ЯРД с твердофазным реактором	1. Сложность отработки (36) 2. Дороговизна делящегося вещества (37) 3. Опасное излучение (38)	Осваиваются; отдельные двигатели проходят стендовые испытания (39)
ЭРД 18 термического типа	1. Малая тяга 34 2. Большая удельная масса 33 3. Зависимость параметров РД от расстояния до Солнца (для СРД) (40)	Осваиваются; проходят стендовые испытания, отдельные РД используются в искусственных спутниках Земли (41)
СРД		Осваиваются; не применяются (42)
Электростатические РД 23	1. Малая тяга 34 2. Большая удельная масса 33 3. Малая величина ускорения, которое приобретает ракетный аппарат при работе РД (43)	Осваиваются; проходят стендовые испытания; отдельные РД используются в искусственных спутниках Земли и космических аппаратах (44)
24 Электромагнитные РД		

Key: (1). Type of RD. (2). Specific mass. (3). Special features/peculiarities of RD, which are determining effectiveness of their use/application on rocket apparatus. (4). N•s. (5). kg. (6). kg•s. (7). W. (8). n. (9). kgf. (10). Possibility of changing thrust over a wide range. (11). Simplicity of construction/design. (12). Possibility of changing thrust. (13). Possibility of designing of high thrusts. (14). Gas. (15). Extreme simplicity of construction/design. (16). Very high reliability. (17). with solid-phase reactor. (18). thermal type. (19). High values. (20). Possibility of designing of high thrusts. (21). Possibility of continuous operation. (22). Possibility of utilization of working medium/propellant with low value μ. (23). Electrostatic. (24). Electromagnetic. (25). Possibility of continuous (of up to several years) operation. (26). Absence of need in heating of working medium/propellant to high temperature. (27). Special features/peculiarities of RD, which limit their use/application in rocket apparatuses. (28). Degree mastery. (29). Comparatively low value $I_{уд}$ due to small quantity of heat, which is isolated in the presence of chemical reactions, and large value μ of chemical reaction products. (30). They are mastered and extensively are used. (31). They are mastered; they are used rarely. (32). Low value. (33). Large specific mass. (34). Low thrust. (35). They are mastered; they are used in artificial Earth satellites and in space vehicles for

creation of low thrusts. (36). Complexity of adjustment. (37). High costs of fissionable material. (38). Dangerous radiation/emission. (39). They are mastered; separate engines undergo bench tests. (40). Dependence of parameters of RD on solar distance (for SRD). (41). Are mastered; undergo bench tests; separate RD are used in artificial Earth satellites. (42). Being implemented; they are not used. (43). Low value of acceleration, which acquires rocket apparatus with work of RD. (44). Are mastered; undergo bench tests; separate RD are used in artificial Earth satellites and space vehicles.

Page 30.

In contrast to heat exchange of YaRD radioisotope sail, mixture and photon nuclear engines, and also YaRD with exhaust jet, which consists of the flow of nuclear particles, do not need any intermediate accelerating device of exhaust jet: the high speed of nuclear particles, their mixture with the working medium/propellant also of photons is the result of the course of nuclear reaction, and task consists only of giving to exhaust jet the required direction.

Apparatuses can be moved in outer space, using so-called solar sail (Fig. 1.19). It creates effort/force to the apparatus as a result of the reflection of sunbeams by the surface of thin gauge sheet (sail). In order to obtain the greatest thrust, the surface of

sail must be oriented in the sun (it is perpendicular to the direction of solar rays/beams).

One should emphasize that are developed for not all types examined above of rocket engines. For example, are not yet created thermonuclear engines, since is not solved the problem of control of thermonuclear fusion. Short comparative rocket-motor characteristic is given in Table 1.1.

Other jet engines can be subdivided as follows.

1. According to type of environment, utilized as working medium/propellant:

a) jet engines;

b) hydrojet engines which develop thrust by accelerating water in their channels and can work in water and on surface of water;

c) jet engines, which use as working medium/propellant atmosphere of other planets.

2. According to type of utilized energy: a) chemical and b) nuclear.

From the engines indicated wide development underwent only VRD. The energy source for the work of VRD is, as has already been indicated, the chemical reaction or burning in which the fuel (e.g., kerosene) interacts with the atmospheric oxygen (oxidizer).

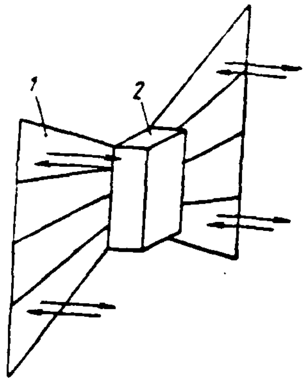


Fig. 1.18. The solar sail: 1 - expanded/scanned sail; 2 - payload.

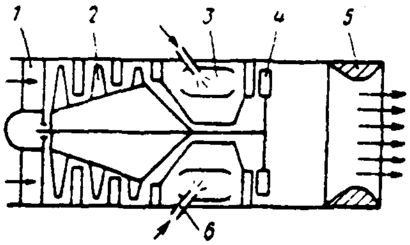


Fig. 1.19.

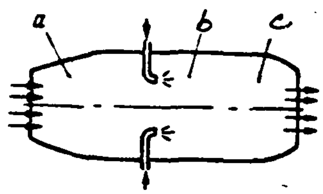


Fig. 1.20.

Fig. 1.19. Diagram of turbojet engine: 1 - diffuser; 2 - multistage axial-flow compressor; 3 - annular combustion chamber; 4 - turbine; 5 - jet nozzle; 6 - injector of fuel injection.

Fig. 1.20. Diagram of compressorless (direct-flow/ramjet) VRD: a - diffuser; b - combustion chamber; c - jet nozzle.

Page 31.

Nitrogen, which is contained in the air, does not accept

participation in the reaction of burning, but it is heated by the heat, which is isolated in the reaction process, and together with other reaction gases flows out from the engine. Due to the presence of nitrogen in the air the temperature in the chamber of a VRD is significantly lower than in the chamber of thermal rocket engines.

Air density with an increase in altitude above the surface of the Earth considerably is decreased. Therefore VRD can work only to the height/altitude of 25-30 km.

VRD are subdivided using the method of air compression before its supply into the combustion chamber to the compressor ones and the compressorless ones.

In compressor of VRD the air is compressed by the special aggregate, called compressor. Compressor (axial or centrifugal) is rotated by the gas turbine which is established/installed on one shaft with the compressor (Fig. 1.19). Turbine works on the reaction products of reaction of fuel and air. Such engines are used for the flights at a velocity, which does not exceed the local velocity of sound at the given height/altitude more than four times, i.e., to M=4.

Compressorless VRD, called also direct-flow/ramjet, are used at the higher flight speed (M=5-10). The air, which encounters to the engine, is compressed into its diffusor (Fig. 1.20) as a result of

the braking, with which the air speed is decreased, and its pressure and temperature increase/grow.

Chapter II.

General Information About Rocket Apparatuses.

§2.1. Fields of application of rocket apparatuses.

Rocket engines in the overwhelming majority of the cases are intended for the high-speed flight vehicles, which include the guided ballistic missiles, AA guided missiles (ZUP) and antimissile missiles, rocket-carriers, space and other rocket vehicles.

The guided ballistic missiles include the rockets, which the significant part of the trajectory after engine cutoff move over the inertia. For example, if we disregard/neglect the effect/action of the gravitational poles of the sun and moon, then in the gravitational field of the Earth rocket moves over the curve, which is the part of the ellipse (Fig. 2.1); this curve they call ballistic.

On the place of start ballistic missiles are subdivided into the ground-based ones and the ship ones, while on the firing distance to the rockets of low, medium and large distance.

Army rockets are moved together with the troop formations and participate in their combat operations/processes.

Strategic missiles are of medium distance and large distance; latter/last rockets call also intercontinental. The starting/launching of strategic missiles can be accomplished/realized on the commands/crews from the single center in the interests of waging war as a whole.

Page 32.

ZUR are used for the destruction of the aircraft of the enemy and of antimissile missile - for organizing the anti-ballistic missle system (PRO), i.e., for the protection from the rockets of enemy.

Carrier rockets serve for the starting/launching of artificial Earth satellites (ISZ) and other space vehicles (KA). These rockets according to device and operating principle are similar to the strategic missiles and can differ from them in terms of the nose section (payload), the flight trajectory, etc.

By space vehicles are ISZ apparatuses for the flights to the moon and the planets of the solar system. KA with the crew aboard, i.e., manned KA, are called spacecraft. Are published scientific

projects of KA for the interstellar flights, in particular for the flights of one planetary system into another [1].

There is a large number of other types of the rockets of combat and peaceful designation/purpose, including antitank missiles, aircraft rockets for the firing at the aircraft, the meteorological investigation, hail-dispersing rockets, etc.

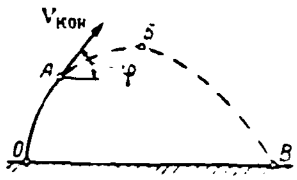


Fig. 2.1. The flight trajectory of the single-stage ballistic rocket: O - launching point; A - cutoff point; Б - the highest point in the trajectory; В - impact point; OA - powered flight trajectory; AВ - inactive leg.

§2.2. Designation/purpose of the engines of rocket apparatuses.

The engines of rocket apparatuses according to the designation/purpose are subdivided into the march ones and the helmsmen. Sustainer engines work basic time and communicate to apparatus the required total impulse. Vernier engines serve for the path control of the motion of apparatus.

KA (including ISZ) have aboard the rocket engines of different designation/purpose, including corrective, braking engines, and also orientation system engines and stabilization of position of KA.

The corrective engines use for the correction (correction) trajectories of KA during its flight with the switched-off sustainer engine.

Braking engines provide braking of KA (ISZ), for example, for the trimmings of ISZ from the orbit, for the creation of the artificial satellite of moon or planet, for the landing of KA on the moon and the planets, which do not have sufficiently dense atmosphere.

Page 33.

The orientation system engines accomplish/realize the turn KA relative to one or the other axis/axle, necessary for its orientation in outer space before firing of the brake and corrective engine, for orienting the antenna of KA to the ground or panels of the solar panel in the sun, during the mating two KA in orbit, and so forth.

The engines of stabilization system prevent rotation or angular oscillations of KA relative to one or the other its axis/axle.

Fig. 2.2 depicts general view of automatic interplanetary space station (AMS) "Venera-4", launched on 12 June 1967 and for the first time in the history of humanity of that carried out a smooth chute and the direct measurements in the atmosphere of another planet. Together with the instruments and the equipment into composition of

KA "Venera-4" entered the corrective engine installation and the low-thrust engines (micromotors) of orientation system.

Rocket engines, in particular ZhRD [liquid propellant rocket engine], can be used also as main engine of aircraft. Such aircraft with the crew are aboard called boost-glide vehicles. The boost-glide vehicles, capable of carrying out an orbital flight of ISZ with the subsequent landing, call space aircraft. They can find use for servicing of orbital spacecraft (space stations): for the delivery/procurement of load, replacement of crew and i.e.

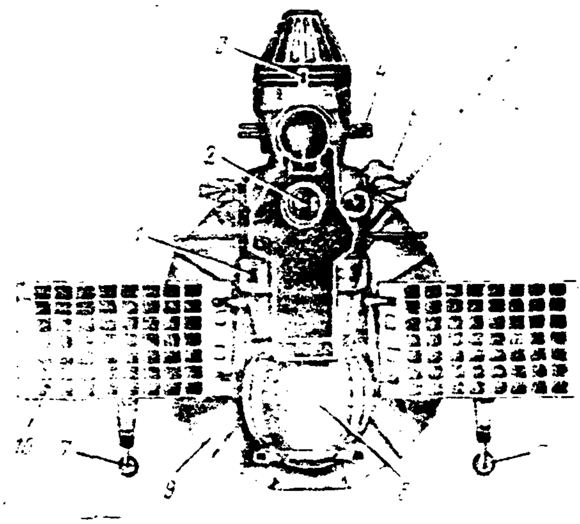


Fig. 2.2. General view of AMS "Venera-4": 1 - orbital compartment; 2 - orientation sensor to the star; 3 - corrective engine installation; 4 - micromotors of orientation system; 5 - counter for the cosmic-ray research; 6 - sensor and the rod of magnetometer; 7 - semidirectional antenna; 8 - launch module; 9 - highly directional parabolic antenna; 10 - solar panel.

Page 34.

Rocket engines (ZhRD or RDTT [solid-propellant rocket engine]) can be the pilot engines of aircraft with VRD for the decrease of the aircraft take-off roll, for a short-time increase in the velocity of flight, etc.

§2.3. Mass characteristics of rocket.

The typical schematic of single-stage rocket with ZhRD, which work on the bipropellant, is depicted in Fig. 2.3.

Section with the payload is usually located in the head section (nose section) of rocket.

Besides first, in the composition of rocket it is possible to isolate instrument, tank and power bays. In the instrument compartment is placed the equipment for the system of control (SU) of rocket flight and of the work of engine. The basic volume of rocket occupies the tank compartment (section with the tanks of propellant components). In the engine, or the weakest, section is placed the engine. Power bay can be ended by end ring (frame/former), on which are assembled such actuating elements of SU as jet vanes or vernier engines. Furthermore, to the frame/former indicated are fastened power packs for the installation of rocket on the starting/launching (starting) device.

Let us examine the basic mass parameters of rocket.

The initial mass of rocket $m_{\text{нач}}$ call mass rockets at the moment of its start of relatively fixed or mobile launcher, i.e., at the

moment of start. Therefore the initial mass of rocket is called also starting.

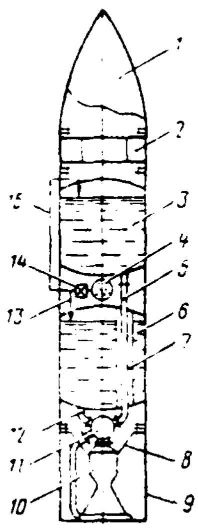


Fig. 2.3. The typical schematic of single-stage rocket with ZhRD, which work on by double component the fuel/propellant: 1 - nose section with the payload; 2 - instrument instrument package for the control system; 3 - oxidizer tank; 4 - tank/balloon with the compressed gas; 5 - conduit/manifold of the delivery of oxidizer to the pump of turbopump unit (TNA); 6 - fuel tank; 7 - tunnel for pipeline; 8 - thrust construction; 9 - weakest (engine) section; 10 - chamber/camera; 11 - TNA; 12 - conduit/manifold of the delivery of fuel to the pump of TNA; 13 - conduit/manifold of the supercharging/pressurization of fuel tank; 14 - pressure reducer of gas; 15 - conduit/manifold of the supercharging/pressurization of oxidizer tank.

Page 35.

The total mass of components of propellant $m_{з.т}$ is their mass, poured (serviced) into the tanks. Into the total mass of propellant components are included:

a) the mass of the propellant components, [illegible] from the moment/torque of firing (launch/starting) [illegible] rocket; let us designate the mass $m_{п.т}$;

b) indicated the mass of working components of propellant $m_{р.т}$, expended with the work of engine from the moment/torque of the start of rocket under the effect/action of the operating engine to that moment/torque when engine ceases to develop thrust;

c) the mass of the remainders/residues of the components of propellant $m_{о.т}$, of those remaining in the tanks and the mains, which connect tanks with the chamber/camera (to the cutoff valves), after engine cutoff.

Consequently, the total mass of propellant components is equal to

$$m_{з.т} = m_{п.т} + m_{р.т} + m_{о.т}.$$

The masses indicated are shown in Fig. 2.4 in the diagram of the

simplest rocket with one-component ZhRD.

The dry mass of rocket $m_{\text{сух}}$ encompasses the mass of payload $m_{\text{п.г}}$, the mass of engine installation (DU) $m_{\text{ду}}$ and the mass of the system of control $m_{\text{су}}$, i.e.,

$$m_{\text{сух}}=m_{\text{п.г}}+m_{\text{ду}}+m_{\text{су}}.$$

Dry mass of DU consists of the mass of engine, tanks, feed system the component of propellant and support systems.

The finite mass of rocket $m_{\text{кон}}$ is its mass after complete engine cutoff (after the moment/torque when engine it ceases to develop thrust).

The initial mass of rocket differs from its mass after servicing of tanks with components of propellant $m_{\text{запр}}$, since

$$m_{\text{запр}}=m_{\text{нач}}-m_{\text{п.т}}.$$

Taking into account the adopted above designations, it is possible to write also the following relationships/ratios, which link different components of mass of rocket:

$$m_{\text{нач}}=m_{\text{сух}}+m_{\text{р.т}}+m_{\text{о.т}};$$

$$m_{\text{нач}}=m_{\text{кон}}+m_{\text{р.т}};\quad m_{\text{кон}}=m_{\text{сух}}+m_{\text{о.т}}.$$

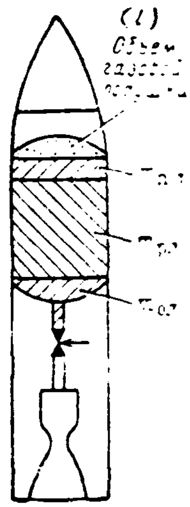


Fig. 2.4. Schematic of rocket with one-component ZhRD.

Key: (1). Volume of gas cushion/pad.

§2.4. The characteristic velocity of rocket.

Multistage rockets.

The universal quantitative characteristic of ballistic missile is its characteristic velocity which is called the speed, necessary for the solution of the ballistic problem: the achievement of the prescribed/assigned distance or height/altitude, starting/launching of satellite to the designated orbit, etc.

Page 36.

Available characteristic velocity V_{xap} is the final speed of rocket $V_{кон}$, which it acquires under the following conditions:

a) in the absence of atmosphere, and consequently, in the absence of the aerodynamic drags and thrust losses which occur in the presence of the atmosphere (see §1.3);

b) in the absence of the forces of the gravitational fields of planets, moon and sun.

Such conditions are provided during finding of rocket in outer space far from the planets and the sun. The speed indicated can be determined according to the equation of Tsiolkovskiy which was by him published into 1903.

The equation of Tsiolkovskiy can be written in the following forms:

$$V_{кон}=W_{э.п}\ln\frac{m_{нач}}{m_{кон}}; \qquad (2.1)$$

$$V_{кон}=W_{э.п}\ln\left(1+\frac{m_{р.т}}{m_{кон}}\right). \qquad (2.2)$$

The equation of Tsiolkovskiy indicates the parameters on which depends the final speed of rocket. The greater the speed $W_{э.п}$ and the relation $m_{нач}/m_{кон}$, the greater the final speed of rocket and the wider than the possibility of its utilization for the solution of different ballistic problems. If we accept value $m_{нач}$ invariable, then relation $m_{нач}/m_{кон}$ can be increased by decreasing the value $m_{кон}$. Consequently, for obtaining the large final speed of rocket under the conditions pointed out above and, consequently, also the large available characteristic velocity during the design of the engine (it is more accurate, engine installation) it is necessary to attain:

a) the large exhaust gas velocity behind the nozzle of engine

chamber W_c; the speed indicated is connected with speed $W_{эп}$ with equation (1.4);

b) the low finite mass of rocket $m_{нач}$ (with the prescribed/assigned mass of payload), i.e., low mass of DU, the system of control and remaining/residues of working medium/propellant (propellant components) in the tanks.

Great attention is paid to an increase in speed W_c. The decrease of the finite mass of rocket $m_{кон}$ and, therefore, an increase in relation $m_{нач}/m_{кон}$ is less effective, since the relation indicated in equation (2.1) will cost under log sign.

During rocket flights in the atmosphere and in the sphere of influence of gravitational fields the final speed of rocket is lower than its characteristic velocity. During the flights indicated it is necessary to select the optimal flight trajectory, i.e., such trajectory with which the total losses of characteristic velocity have small value,

$$V_{хар} - V_{кон} \to \left(\sum \Delta V_{хар}\right)_{min}.$$

Page 37.

For conducting the rocket along the optimal trajectory in its composition must be the system, which creates control forces, for

example jet vanes or vernier engines with the system of their drive.

Value $m_{нач}/m_{кон}$ is determined by the structural/design perfection of rocket and has for each type of rockets one or another limit. The values of speed W_c depend on the type of engine and they also have a limit. Therefore limiting values V_{ch} comprise for the single-stage rockets RDTT of 3000-3500 m/s, for the rockets with ZhRD - 6000-6500 m/s and for the rockets with YaRD - 9000-10000 m/s. Consequently, the ballistic possibilities of single-stage rockets, evaluated by their characteristic velocity, are limited.

For the solution of many ballistic problems are required substantially the higher values of characteristic velocities.

For guiding a satellite into orbit around the Earth by rocket apparatus must be achieved/reached the speed, greater than circular, or orbital velocity; it on the surface of the Earth is equal to 7900 m/s, and at an altitude h=200 km - 7790 m/s.

If rocket apparatus acquires the escape velocity (escape velocity), then it escapes the pull of gravity of the Earth and can be directed toward the moon, to other planets, to sun and so forth escape velocity on the surface of the Earth is equal to 11190 m/s, and at an altitude h=200 km - 11010 m/s.

Due to losses examined above of characteristic velocity necessary characteristic velocity must be accordingly higher: for example, for the injection of ISZ into low circular orbit value V_{xar} is equal to 10000-10200 m/s.

The very effective method of obtaining the high values of characteristic velocity is the use/application of a composite/compound (multistage) rocket. This rocket consists of two and more than steps/stages. Multistage rockets are subdivided into the rockets with the cross and parallel staging.

Let us explain the special features/peculiarities of work of multistage rockets based on the example of two-stage rockets (Fig. 2.5).

With the work of DU of first stage two-stage rocket with the tandem staging (see Fig. 2.5a) it is necessary to consider as one step/stage, which consists of the basic building block of first stage with DU and the nose section, which in this case is the second step/stage, i.e., it projects in the role of payload for the basic building block of first stage. After the consumption of bulk of propellant components in the tanks of DU of first stage its basic

building block is separated/liberated, so that second-stage engines spend their power only on the dispersal/acceleration of the remaining part of the rocket (second step/stage). The second step/stage consists of basic building block and nose section with the payload.

Page 38.

Two-stage rocket with the parallel staging (see Fig. 2.5b) is actually 1 1/2- by stage rocket. From the very beginning of the flight of this rocket they work both the engines of the basic building blocks of first stage and the engine of the block, which is first one of the main blocks of first stage, and after the department/separation of other basic building blocks of first stage - by basic building block of the second step/stage, i.e., after the department/separation of other basic building blocks of first stage in the composition of rocket remain the basic building block indicated and the nose section (payload and section SU).

As is evident, the effectiveness of multistage rockets is explained by the fact that during their utilization it is possible to considerably increase relation $m_{нач}/m_{кон}$, and, consequently, the characteristic velocity of rocket.

A reduction/descent in the mass is least effective at first

stage and it is most effective - on the latter. Therefore to reduction/descent masses of DU and remainders/residues of components of propellant (working medium/propellant) at the latter/last steps/stages pay great attention.

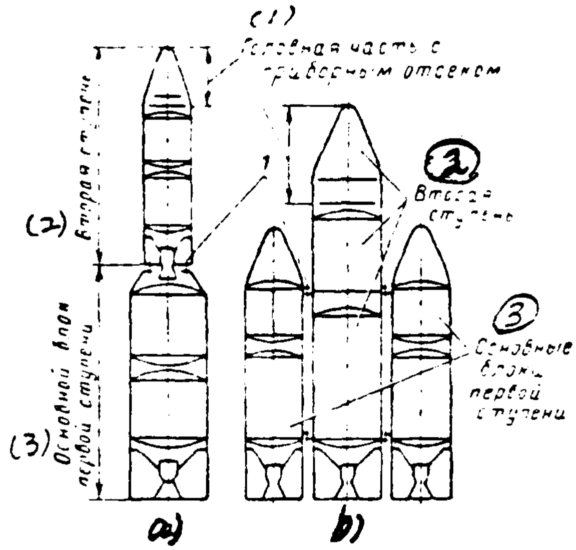


Fig. 2.5. The schematics of the two-stage rockets: a) with the tandem staging; b) with by parallel staging; 1 - explosive bolts.

Key: (1). Nose section with the instrument compartment. (2). Second step/stage. (3). Basic building blocks of first stage.

§2.5. Requirements for the engine installations.

The power plant is effective for use in the rocket vehicle if it gives it a high characteristic velocity for this purpose, the specific impulse of the engine should be sufficiently large, while the mass of the engine is small. A reduction in the mass of DU is achieved at compactness of its construction and by the utilization of working medium with the large density. Effectiveness of DU increases/grows, if the unproductive expenditures of working medium/propellant in the period of engine starting are low, and after its disconnection in the tanks remains only a small quantity of

unused working medium/propellant.

DU must also provide the possibility of changing in thrust and creation of efforts/forces and moments/torques for the trajectory guidance of rocket vehicle. Need in a change in the thrust appears when engine is the actuating element of the system of control of rocket vehicle. As example can serve the control system of the apparent velocity (in abbreviated form system of RKS). The apparent velocity is determined by integrating the apparent acceleration, i.e., the total acceleration, communicated to rocket vehicle by all acting on it forces (engine thrust, the aerodynamic resisting forces of the atmosphere, etc.), with exception of the forces of gravitational fields. The apparent acceleration is measured with the aid of the special instruments (accelerometers), which form part of the control system. In flight of rocket vehicle under conditions of the strong gravitational field of planet the apparent velocity of vehicle can considerably differ from speed relative to launching point by the surface of planet.

The work of system of RKS lies in the fact that at each point of powered flight trajectory the actual value of apparent velocity $V_{\text{кaж.д}}$ is compared with the programmed value of apparent velocity $V_{\text{кaж.пр}}$. Values $V_{\text{кaж.пр}}$ are embedded into the special control-system equipment prior to the start of ballistic missile. In the case

$V_{каж.д} < V_{каж.пр}$ the system РКС gives command/crew to an increase, while in the case $V_{каж.д} > V_{каж.пр}$ — to the decrease of the engine thrust.

The system of the creation of control forces and moments/torques must provide at each point of the powered flight trajectory of ballistic missile least possible deviation of the actual values of the angles of pitch, course and yaw (Fig. 2.6) from their programmed values.

Pitch angle call the angle between longitudinal axis rockets and local horizon at the point of intersection of the surface of planet with the line, which connects the center of mass of rocket with the center of planet.

Pitch angle is arranged/located in range plane, i.e., in trajectory plane of ballistic missile; the plane indicated is passed through the launching points and incidence/drop in the rocket, and also through the center of planet.

The course angle (or yaw) is called the angle between the axis of rocket and range plane. The attitude of roll of rocket is determined by the turning angle of the transverse axes of rocket around its longitudinal axis from the calculated position of the axes/axles indicated.

Page 40).

The pitch angles, course and bank designate by letters ϑ, Ψ and γ respectively. For the suppressing number of ballistic missiles the control system provides conditions [illegible] i.e., the actual values of angles Ψ and γ must as less as possible to differ from zero, and the actual value of angle ϑ also to least possible degree must differ from programmed values, different for each point of powered flight trajectory.

To engine installation must be inherent also the following qualities.

1. High reliability, i.e., provision of reliable efficiency during preset time under prescribed/assigned conditions. Especially high requirements are imposed on reliability of DU of the manned spacecraft.

2. Simplicity of construction/design and diagram. It VI much determines reliability of DU.

3. High manufacturability. The processes of manufacture of parts

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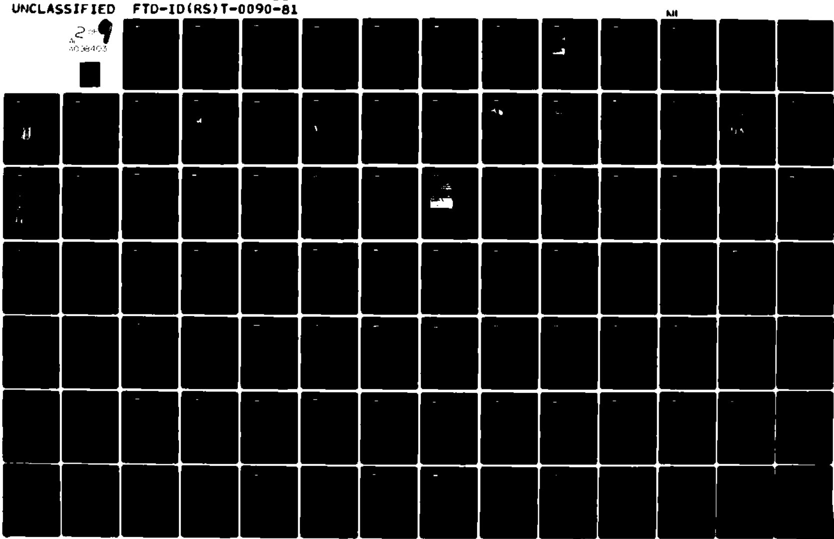

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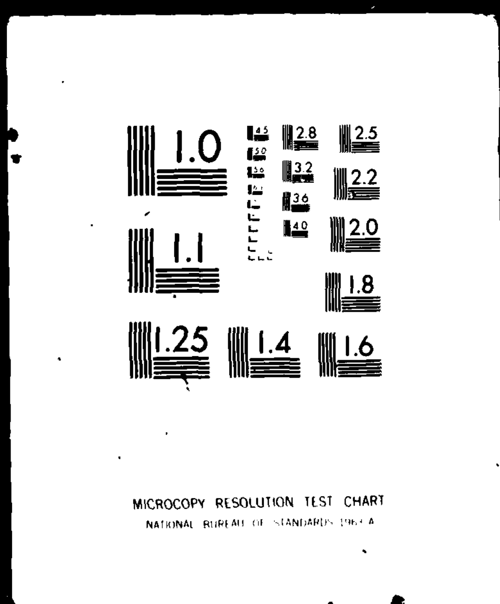

of DU, and also the processes of welding, stick, assembly of assemblies, aggregates and engine as a whole and the control/check of their quality must be sufficiently simple. Furthermore, the processes indicated must yield to automation.

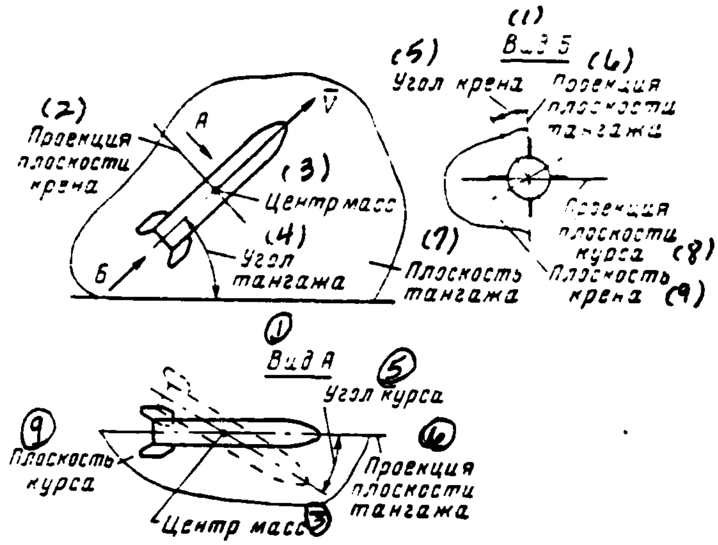

Fig. 2.6. Diagram of the layout of the pitch angles, course and bank.

Key: (1). Form. (2). Projection of rolling plane. (3). Center of mass. (4). Pitch angle. (5). Roll attitude. (6). Projection of plane of pitch. (7). Pitching plane. (8). Projection of plane of course. (9). Rolling plane.

Page 41.

4. Convenience in operation. Here are included:

a) simplicity, cheapness and the safety of transportation, storage, maintenance/servicing and repair;

b) invulnerability during the operation, i.e., insensitivity to the external agencies, for example to the pollution/contamination of propellant components and to different errors of the service personnel, to the effect of conditions of outer space (for example, to the effect of micrometeorites), etc.

7. Smallest cost/value and time of manufacture. This relates to the cost/value of structural materials, fuel/propellant and equipment for the manufacture, the assembly and the testing of DU.

If into composition of DU enter several sustainer engines, then the spread of their thrust with the work in the basic mode/conditions, and also during the starting/launching and the disconnection must be low. Otherwise appears the large perturbing moment/torque, which must be compensated, which leads to the appropriate losses.

DU of combat missiles must allow/assume long shelf-life during the significant oscillations/vibrations of the temperature of the atmosphere, including with the charged/filled tanks. In the case of the emergence of the necessity engine starting and missile takeoff must be provided into possibly the shorter time interval. This requirement requirement has special importance for ZUR, strategic ballistic missiles, which are found on uninterrupted standby alert.

Are very specific requirements for the rocket engines of space vehicles. For KA optimal is the sustainer engine of multipurpose designation/purpose. It must allow/assume the possibility of multiplying, including of afterward prolonged (of up to several years) flight into space under conditions of weightlessness, and the possibility of a significant reduction/descent in the thrust.

The example of multipurpose engine is the sustainer engine of the "Apollo" spacecraft, which is designed for 50 inclusions/connections with the general/common/total duration of operation τ=750 s. This engine it provides:

1) to three trajectory corrections of ship both in flight to the moon and during return flight;

2) braking ship during the approach to the moon for the injection into orbit of the satellite of the moon (τ=360 s);

3) a change in the original elliptic orbit during the motion around the moon to the circular (τ=360 s);

4) start with the orbit of the satellite of the moon for the

flight to the Earth (τ=150 s), and also some other operations/processes.

An error in thrust impulse/momentum/pulse, created brake and correcting by engines of KA, must be low for the safeguard of the prescribed/assigned calculated final speed. The deviation of the actual value of this speed from the calculated must be small (in certain cases not more than +-0.005 m/s).

The construction/design of the engines, intended for the landing of KA on the surface of other planets, must allow/assume conducting chemical and thermal sterilization for the exception/elimination of the possibility of the recording of terrestrial microorganisms to other planets.

Page 42.

§ 2.6. Short history of rocket engines.

The first rocket vehicles were the solid-propellent rockets, invented in ancient China. The first in the world scientific design of rocket vehicle for the manned flight proposed Russian revolutionary, member of "Narodnaya volya", inventor N. I. Kibal'chich (1853-1881). Being located in the prison conclusion in Petersburg for the participation in the attempt on the tsar, Kibal'chich during March 1881 developed the "Design of aeronautical instrument", in which were examined the device of solid propellant engine, control of rocket vehicle by changing the angle of the slope of engine, programmed combustion behavior, the stabilization of vehicle and other questions.

The bases of the theory of cosmonautics and rocket engineering placed brilliant Russian Soviet scientist-inventor - pioneer of rocket engineering and cosmonautics K. E. Tsiolkovskiy (1857-1935). In 1903 appeared his classical work "investigation of outer space by

jet drives", in which Tsiolkovskiy from the engineering positions for the first time derived the laws of the motion of rocket as the bodies of variable mass, based the possibility of the utilization of rockets for the interplanetary flights, were indicated the rational ways of the development of missile construction and cosmonautics.

For the first time in the world Tsiolkovskiy gave principles of the theory of liquid-propellant rocket engines and were indicated the elements of their construction/design. By it they were examined and recommended to the utilization different fuels/propellants for ZhRD. As the oxidizers it planned to use liquid oxygen, ozone, nitrogen pentoxide, while as the fuel - liquid hydrogen, methane, hydrocarbons, benzene, gasoline, turpentine and other substances.

Constantine Eduardovich Tsiolkovskiy.

Page 43.

In the work, published in 1911 [45, pg. 89-90], Tsiolkovskiy indicated the advantages of the utilization of nuclear energy in the rocket engine. In the same work Tsiolkovskiy wrote: "It can be, with the aid of the electricity it will be possible to in the course of time with give enormous speed to the ejected from the jet drive particles.".

First ZhRD (although low sizes/dimensions) were created and tested American they were scientific, one of the pioneers of rocket technicians by R. Goddard (1882-1945).

Experiments with the oxygen-hydrocarbon liquid propellant Goddard beginnings in 1920, and the bench tests of ZhRD on the oxygen-ether/ester fuel/propellant - in 1921, 16 March [redacted] 1926 in Wooster Goddard produced the first starting/launching of the rocket with ZhRD (fuel-liquid oxygen and gasoline), which reached the altitude of 12.5 m; the flight lasted 2.5 s.

It is earlier, in 1919, Goddard published the book "Method of achieving the extreme heights/altitudes", in which he presented theory and design concepts of multistage rockets. In 1923 the questions indicated were examined by German scientist, one of the founders of rocket engineering and cosmonautics by G. Oberth (born in 1894). In 1929 Tsiolkovskiy published the work "Space multi stage rockets", in which he developed diagrams and theory of multistage rockets [45].

One of the pioneers of rocket engineering is Yu. V. Kondratyuk (1897-1942). In 1919 in the work "that, who will read in order to build" it it developed number of the basic problems, connected with the space flights. In the theoretical studies "Conquest of

interplanetary spaces", published in 1929 in Novosibirsk, Kondratyuk, without knowing about the works of Tsiolkovskiy, originally derived the fundamental equations of motion of rocket. In Kondratyuk's works are examined the theory of multistage rockets, the economical landing of rockets on the planet with the utilization of atmospheric braking; Kondratyuk proposed flights to the moon and the planets with the injection into orbit of their artificial satellite and the subsequent department/separation of takeoff and landing spacecraft as energetically advantageous. Furthermore, by them are proposed as the fuels for the rocket engines some metals, metalloids and their hydrogen compounds, for example hydroboride [20].

Large representative of the Soviet school of missile construction was scientist and inventor F. A. Tsander [1887-1933). He as early as the student years studied the transactions of Tsiolkovskiy and he was interested in questions of space flights.

Page 44.

In 1924 Tsander published in the journal "Technology and life" the article "Flights/passages to other planets", in which he presented its basic idea - the combination of rocket with the aircraft for the takeoff from the Earth with the subsequent combustion of those placed by the unnecessary ones of metallic parts (for example, the wings of

aircraft and tanks after the production/consumption/generation of propellant components) in the thrust chamber for an increase in the flying range of rocket [44]. Since 1924 Tsander studied the idea of solar sail, for the first time expressed by B. Krasnogorskiy in the novel "on the waves of ether/ester" (1913).

In 1930-1931 Tsander constructed and tested in the compressed air with the gasoline the first jet engine OR-1, which developed thrust from 1420 mN [145 gf] and more.

One of the pioneers of rocket engineering, who made the basic contribution to the development of Soviet engine construction, is academician V. P. Glushko (it was born in 1908). Beginning with 1924 Glushko it publishes the series/number of popular science and scientific works on the cosmonautics.

On 15 May, 1929, on Glushko's proposition in the gas-dynamic laboratory (GDL) in Leningrad was organized first group, and then section, which began the creation of electrical and liquid propellant rocket engines.

Glushko - builder of the first in the measure electrothermal rocket engine (Fig. 2.7a) which was by it theoretically developed, constructed and tested in GDL in 1929-1933. Engine consisted of

chamber/camera with the nozzle, into which with special mechanism were supplied the metallic wires, which simultaneously served as the conductor of electric current and as working medium/propellant. Wires exploded with the electric current (pulsed discharges of capacitor/condenser). The forming vapors of metal flowed out behind the nozzle, creating thrust. Instead of the wires into the chamber/camera could be supplied through injector-electrodes the streams of the electro-conductive liquids (mercury, electrolytes), which also exploded in the chamber/camera with electric current.

In 1930-1931 under the management/manual of Glushko in GDL was developed and constructed first experimental ZhRD ORM-1 (experimental rocket motor) (see Fig. 2.7b). It tested on nitrogen tetroxide and toluene, and also on liquid oxygen and gasoline, moreover in the latter case it developed thrust to 196 n [20 kgf].

In 1930 Glushko he proposed also subsequently investigated as the components of propellant ZhRD nitric acid, the solutions of nitrogen tetroxide (nitric tetroxide) in the nitric acid, tetranitromethane, peroxide of hydrogen, perchloric acid, berryllium, three-component fuel/propellant - oxygen with hydrogen and beryllium, powders with the dispersed beryllium. In Glushko's the same year were developed special (shaped) nozzle and heat-insulation coatings from zirconium dioxides, applied to the internal chamber wall.

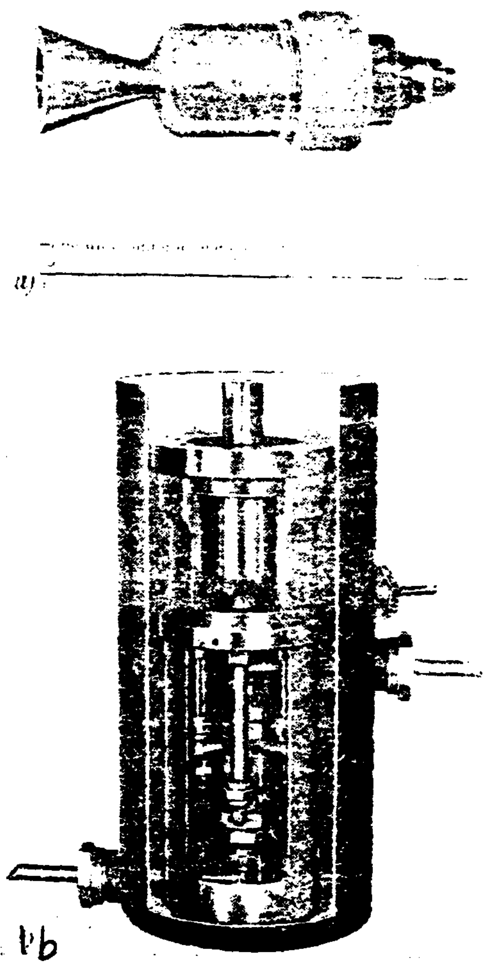

Fig. 2.7. Engines, V. P. Glushkc's created in GDL: a) thermal type first in the world ERD (1929-1933): b) first Soviet ZhRD ORM-1 by thrust 196 n [20 kgf] (1930-1931).

Page 46.

In 1930-1933 in GDL was created the family of ZhRD: from ORM, ORM-1 to ORM-52.

During September 1931 in Moscow with the central council of Osoaviaxim of the USSR was created the group of the study of reactive motion (GIRD (ГИРД) — Group for the Study of Jet Propulsion)) - the public organization, which combined the enthusiasts of rocketry.

During June 1932 the solution of the presidium of the central council of Osoaviaxim in Moscow created the group of the study of reactive motion (GIRD) - scientific research and experimental design organization for the development of rockets and engines. Chief of GIRD assigned S. P. Korolev (1906-1966), who made the basic contribution to the practical cosmonautics.

In GIRD the crew of Tsander developed the design of engine installation with ZhRD OR-2 for boost-glide vehicle PR-1 (fuel-liquid oxygen and gasoline; design thrust - 490 n [50 kgf]).

On 17 August, 1933, on the polygon/range in Nakhabino in the environs of Moscow was launched created by crew M. K. Tikhonravov rocket "GIRD-09" (Fig. 2.8) - the first Soviet experimental rocket. Her engine worked on liquid oxygen, supplied to the chamber/camera with vapor pressure of its own and the solidified gasoline which was

placed in the combustion chamber, developed thrust 245-324 n [25-33 kgf]. During the maiden flight the rocket "GIRD-09" achieved height/altitude of approximately 400 m.

First ZhRD of the construction/design of Tsander (OR-2) was prepared in GIRD in December of 1932. The pupils of Tsander, after his death continuing the matter, initiated in him, reconstructed engine OR-2 - replaced gasoline with ethyl alcohol for reducing/descending the temperature of gases and facilitation of cooling; introduced ceramic lining chambers/cameras and appropriated to it index 02 [23].

Tsander developed/processed in GIRD from January 1933 also the engine 10-ZhRD, which had to work on liquid oxygen and gasoline. During August 1933 was changed the construction/design of engine and gasoline was substituted by alcohol. On 25 November, 1933, took place the first launching/starting of created GIRD under Korolev's management/manual rocket "GIRD-X" with engine 10.

With the resolution of the council of work and defense of the USSR at the end of October of 1933 in Moscow on the base GDL GIRD was created the first in the world reactive/reagent scientific research institute (RNII [(РНИИ). - Scientific Research Institute of Jet Propulsion]).

Sergey Pavlovich Korolev.

Page 47.

The chief of RNII was assigned I. T. Kleymenov (1898-1938), assistant - S. P. Korolev, and from January 1934 - G. E. Langemak (1898-1938). The collective of RNII worked on all basic problems of rocket engineering, maintaining close connection with Tsiolkovskiy.

The collective of the specialists in ZhRD, which grew in GDL, developed in 1934-1938 in RNII a series of experimental engines from ORM-53 to ORM-102 for the work on nitric acid and tetranitromethane as the oxidizers and the first Soviet gas generator GG-1 on the nitric acid with the kerosene and the water, that produced by hours

inert pure gas.

ZhRD ORM-65 (Fig. 2.9), developed in the RNII and which passed official tests in 1936, was the most highly developed engine of that time. It worked on the nitric acid and the kerosene and provided a change in the thrust from 490 n [50 kgf] to 1716 n [175 kgf]. Specific jet firing composed 2059-2109 N•s/kg [210-215 kg•s/kg]. Engine allowed/assumed both manual and automatic starting/launching and were withstood/maintained repeated (to 50) bench tests with the general/common/total duration to 30 min.

The collective of RNII created the series/number of experimental ballistic and winged missiles and engines to them. In 1934-1938 were modern the flights of many rockets. In 1939 were carried out the flight tests of the cruise missile of 212 constructions/designs of Korolev with engine ORM-65, while in 1937-1938 - ground tests of rocket glider RP-318 of the construction/design of Korolev with the same engine. On 28 February, 1940, pilot V. P. Fedorov completed flight on this rocket glider with engine RDA-1-150.

In 1942 took place pilot G. Ya. Bakhchivandzhi's flight on the first Soviet rocket aircraft BI-1, on which was established/installed that developed in the RNII by L. S. Dushkin ZhRD D-1-A-1100 (fuel/propellant - nitric acid and kerosene; rated thrust 10788 n [1100 kgf]).

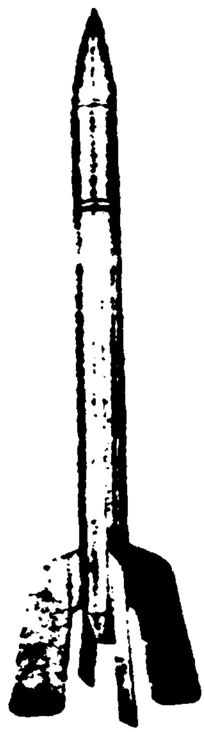

Fig. 2.8. Rocket "GIRD-09".

Page 48.

Job superintendent on ZhRD in GDL Glushko continued this work in the RNII (1934-1938). During this period appeared of Glushko's book the "rockets, their device and use/application" (together with Langemak, 1935) [24] and "liquid propellant for the jet engines" (1936) [12].

In 1939 was created the independent organization, which grew in 1941 into experimental design bureau (OKB) of ZhRD.

In the 40's OKB developed the family of aircraft ZhRD (RD-1, RD-1KhZ, RD-2, RD-3) with the pump feed (fuel/propellant - nitric acid and kerosene). Engines provided multiplyings with a change in the thrust in the Earth from 2743 n [300 kgf] to 8829 n [900 kgf]. Engines RD-1 and RD-1KhZ underwent bench finishing and official tests, and in 1943-1946 - about 400 ground and flight tests on aircraft Pe-2 of the construction/design of V. M. Petlyakov, La-7R and 120R the construction/design of S. A. Lavochkin, Yak-3 the construction/design of A. S. Yakovlev, Su-6 and Su-7 the construction/design of P. O. Sukhoy.

Qualitatively new development stage of rocket engineering was begun after the Second World War in the USA and, especially, in the USSR.

Soviet specialists, beginning with 1947, developed several ten types of powerful/thick ZhRD, which had extensive application on the rockets of different designation/purpose.

To a number of these engines they relate ZhRD RD-107 (Fig. 2.10) and RD-108. They were established/installed respectively to first and second stage of the rocket, with the aid of which was launched on 4 October, 1957, the first in the world artificial Earth satellite. By this starting/launching was established the beginning of the space age in humanity's history.

ZhRD RD-107 and RD-108 were applied also in the three-stage carrier rocket "east" (Fig. 2.11).

On 12 April, 1961, with the aid of the carrier rocket "east" was realized the first in the world flight of the citizen of the USSR YU. A. Gagarin aboard the "Vostok" spacecraft on the orbit of artificial Earth satellite.

Valentina Petrovich Glushko.

Page 43. ZhRD RD-214 and RD-119 (Fig. 2.12), developed in 1952-1957 and 1958-1962 respectively, are established/installed to the carrier rocket "Kosmos". With the aid of this rocket is launched a large quantity of investigation satellites.

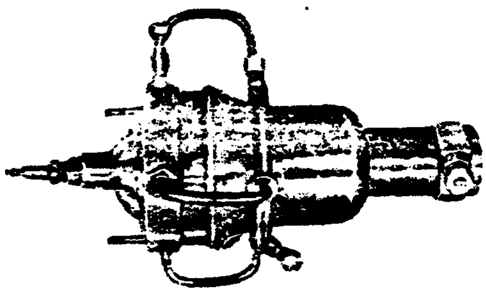

Fig. 2.9. ZhRD ORM-65 (GDL; engine underwent official tests in 1936).

Basic data of ZhRD RD-107, RD-108, RD-214 and RD-119 are brought to the given below table.

ЖРД	(1) Ракета-носитель	(2) Компоненты топлива (3) Окислитель	(4) Горючее	I (5) н·сек/кг (8)	(6) кГ·сек/кг (8)	[illegible] (7) кн	T
РД-107	„Восток", первая ступень (9)	Жидкий кислород (10)	Керосин (11)	3079	314	1000	102
РД-108	„Восток", вторая ступень (12)	То же (13)	.	3089	315	941	[illegible]
РД-214	„Космос", первая ступень (14)	Азотно-кислотный (15)	Продукты переработки керосина (16)	2589*	264*	726	74
РД-119	(17) „Космос", вторая ступень	(18) Жидкий кислород	(19) Несимметричный диметилгидразин	3452*	352*	108	11

Key: (1). Rocket is carrier. (2). Propellant components. (3). Oxidizer. (4). Combustible. (5). N•s. (6). kg•s. (7). kn. (8). kg. (9). "East", first stage. (10). Liquid oxygen. (11). Kerosene. (12). "East" second step/stage. (13). The same. (14). "Kosmos" first stage. (15). Nitrogen-oxygen. (16). Converted products of kerosene. (17). "Kosmos" second step/stage. (18). Liquid oxygen. (19). Unsymmetrical dimethyl hydrazine.

FOOTNOTE [1]. Highest specific impulse in the vacuum among the known engines of this class, which work on the nitric-acid oxidizer and the hydrocarbon fuel.

------ [1]. Highest specific impulse in the vacuum for the oxygen engines of those using the high-boiling fuel.

Page 50.

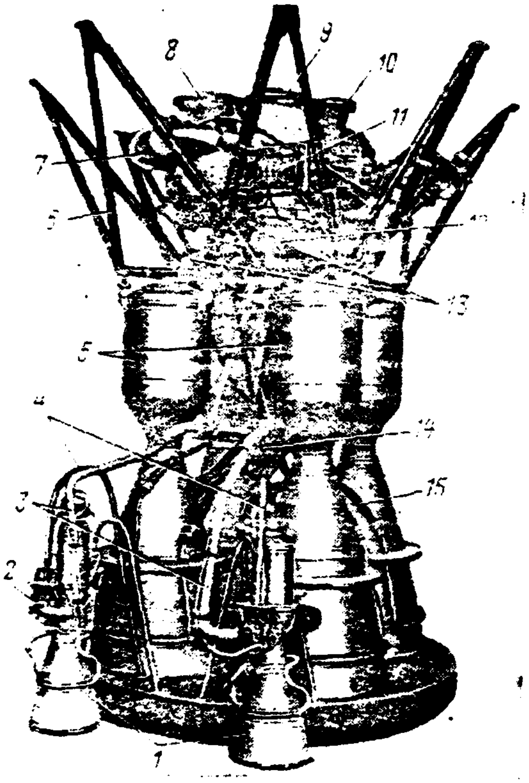

Fig. 2.10. Soviet oxygen-kerosene ZhRD RD-107 "Vostok" with thrust P_n=1000 by kn [102 T] and specific impulse $I_{y\partial.n}$=3079 N•s/kg [314 kg•s/kg] (1954-1957): 1 - steering chambers/cameras; 2 - unit of fluctuation and delivery of oxidizer; 3 - conduits/manifolds of oxidizing agent of steering chambers/cameras; 4 - simulated brackets (constructions/designs of engine are absent); 5 - basic chambers/cameras; 6 - power frame; 7 - gas generator; 8 - housing of heat exchanger on turbine; 9 - intake pipe of pump of oxidizer; 10 -

intake pipe of fuel pump; 11 - pressure sensor in combustion chamber; 12 - main valve of oxidizer; 13 - oxidizer pipes; 14 - main valve of fuel; 15 - fuel pipes.

Page 51.

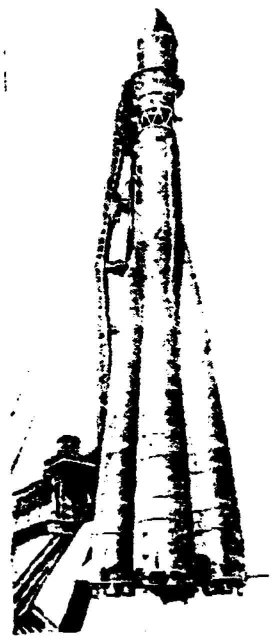

Fig. 2.11. Spacecraft launch vehicle "East".

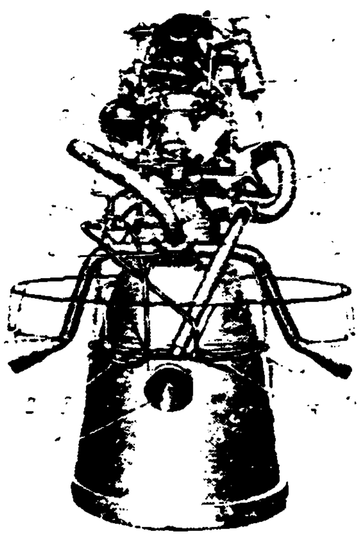

Fig. 2.12. Soviet oxygen-dimethylhydrazine ZhRD RD-119 "Kosmos" with

thrust P_n = 108 kn [11 T] and specific impulse I = 3452 N•s/kg [352 kg•s/kg] (1958-1962): 1 - steering nozzles of pitch (second nozzle from opposite side); 2, 13 - steering nozzles of course; 3, 15 - steering nozzles of bank (second pair of nozzles from opposite side); 4,5,11 - gas distributors with electric drive; 6 - combustion chamber; 7 - cylinder for compressed air; 8 - turbopump unit; 9 - gas generator; 10 - power frame; 12 - mounting ring of the steering system (in the construction/design of engine it is absent); 14 - removable silencer/plug.

Page 52.

On 16 July, 1965, was produced the starting/launching of artificial Earth satellite "Proton 1". The carrier rocket of system "Proton" is capable of bearing several times the large mass of payload, than carrier rocket "East".

On the carrier rocket of system "Proton" are used newly developed powerful/thick highly efficient ZhRD, made in the most modern diagram and which differ in terms of low dimensions and high reliability.

In connection with 40- year-anniversary GDL and grown from it experimental design bureau (1929-1969) in Leningrad on the buildings

of main shipyard and Icannovskiy ravelin of the Petropavlovsk stability where in 1932-1933 was placed gas-dynamic laboratory, were established/installed memorial panels (Fig. 2.13).

Development of missile-space technology in the USSR led to the creation of the new creative collectives whose specialists developed a whole series of intercontinental rockets, carrier rockets, space vehicles and ships.

The basic stages of the mastery/adoption of outer space are reflected in the given below table.

Page 53.

Цель космического полета (1)	Дата запуска ракеты (2)	
	в СССР	в США
Запуск первого ИСЗ (3)	4.10.57	1.2.58
Запуск первого искусственного спутника Солнца (4)	2.1.59	3.3.59
Впервые КА достиг Луны (5)	12.9.59	23.4.62
Запуск первого КА в сторону Венеры (6)	12.2.61	27.8.62
Первый полет человека на космическом корабле по орбите ИСЗ (7)	12.4.61	20.2.62
Запуск первого КА в сторону Марса (8)	1.11.62	28 [illegible]
Выход космонавта в открытое космическое пространство (9)	18.3.65	3.6.65
Впервые КА достиг Венеры (10)	16.11.65	Не осуществлен (10a)
Первая мягкая посадка КА на Луну (11)	31.1.66	30.5.66
Запуск первого искусственного спутника Луны (12)	31.3.66	10.8.66
Впервые КА произвел непосредственные измерения в атмосфере Венеры (13)	12.6.67	Не осуществлен (10a)
Первый облет Луны космическим кораблем и его возвращение на Землю (14)	15.9.68[1]	21.12.68
Первая стыковка пилотируемых кораблей и переход космонавтов из корабля в корабль (15)	16.1.69	7.3.69
Впервые КА совершил посадку на Венеру и передавал с ее поверхности информацию (16)	17.8.70	
Мягкая посадка корабля на Луну с возвращением на Землю (17)	Сентябрь 1970 г.[4] (18)	Июль 1969 г. (19)
Посадка на Луну автоматического самоходного аппарата (лунохода) (20)	10.11.70	Не осуществлен (10a)

Key: (1). Target of space flight. (2). Date of launching of rocket. (3). Starting/launching of first ISZ [- artificial earth satellite]. (4). Starting/launching of first artificial satellite of sun. (5). For the first time KA reached the Moon. (6). Starting/launching first KA to side of Venus. (7). Maiden flight of man aboard spacecraft on orbit ISZ. (8). Starting/launching first KA to side of Mars. (9). Cosmonaut's exit into open outer space. (10). For the first time KA reached Venus. (10a). It is not realized. (11). First soft landing of KA on moon. (12). Starting/launching of first artificial satellite of moon. (13). For the first time KA took direct measurements in the atmosphere of Venus. (14). First flight around of moon by spacecraft and its return to the earth. (15). First manned spacecraft docking and transition/transfer of cosmonauts from to. (16). For the first time KA completed landing on Venus and transmitted from its surface information. (17). Soft landing of ship on moon with return to the earth. (18). September of 1970. (19). July 1969. (20). Landing on moon automatic self-propelled vehicle (Lunokhod).

FOOTNOTE [1]. Automatic station the "Zond-5". ENDFOOTNOTE.

FOOTNOTE [2]. Manned spacecraft "Apollo-8" (with three cosmonauts

aboard). ENDFOOTNOTE.

FOOTNOTE 3. Date of mating. ENDFOOTNOTE.

FOOTNOTE 4. Automatic station "Luna-16", which ensured for the first time in humanity's history the automatic sampling of moon ground and its delivery/procurement to the earth. ENDFOOTNOTE.

FOOTNOTE 5. Manned ship "Apollo-11", which ensured the debarkation of the american astronauts Armstrong and Aldrin 20 July 1969 to the surface of the moon. ENDFOOTNOTE.

FOOTNOTE 6. Automatic station "Luna-17", which ensured for the first time in the world the delivery/procurement of self-propelled vehicle "Lunokhod-1" to the surface of the moon.

Page 54.

The successful completion of the programs of automatic stations "Luna-16" and "Luna-17" indicates the beginning of qualitatively new stage in the cosmonautics - stage of accomplishing extremely complex space experiments with the aid of the automatic machines. Automatic stations are very effective: fulfilling in essence the same functions, as the manned ships, such stations they have smaller

cost/value, they can work greater time, also, under more severe conditions than the manned spacecraft, and furthermore, during the utilization of automatic stations there is no need for to risk cosmonauts' life.

Fig. 2.13. Memorial panel (bronze), established/installed on the building of the Ioannovskiy ravelin of Petropavlovsk stability in Leningrad.

Page 55.

Chapter III.

WORKING MEDIUM/PROPELLANTS OF ROCKET ENGINES.

§ 1.3. General characteristics and the classification of working medium/propellants.

Concept "Working medium of RD" is very wide. Working medium/propellant of RD is called the substance (in this or another state of aggregation) which is used for the creation of reacting force, for the ingress of heat, for the drive of turbine and other engine accessories, etc.

According to their designation/purpose working medium/propellants of RD are subdivided into the bases, the starting/launching ones and the additional ones (Fig. 3.1).

Basic are called the working medium/propellants on which RD works basic time, creating thrust by the rejection of exhaust jet.

Exhaust jet is the flow of working medium/propellant or reaction

products of initial working medium/propellants, moreover substance in exhaust jet by its state of aggregation, chemical composition and parameters usually differs significantly from initial working medium/propellants, which are located in the tanks or the chamber/camera of RD.

Starting/launching working medium/propellants are used during the starting/launching of RD (in the initial period). They are necessary; for example, for the starting/launching chemical RD, if basic propellant components are not capable of independently entering into the reaction.

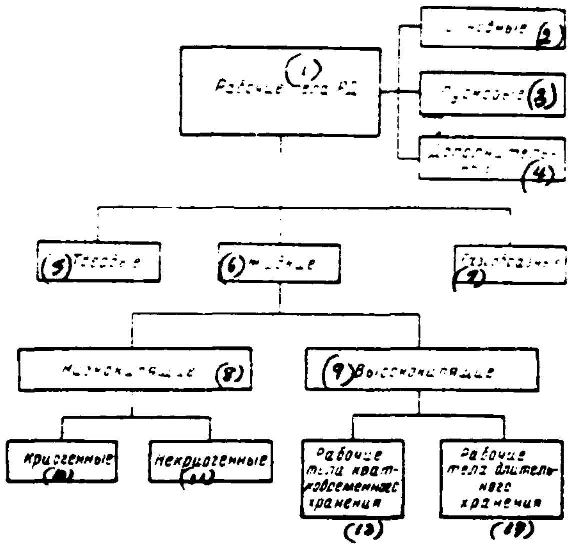

Fig. 3.1. General/common/total classification of working medium/propellants of RD.

Key: (1). working medium/propellants of RD. (2). Basic. (3). Starting/launching. (4). Additional. (5). Solid. (6). Liquid. (7). Gaseous. (8). Being low-boiling. (9). high-boiling. (10). Cryogenic. (11). Noncryogenic. (12). Working medium/propellants of short-time storage. (13). Working medium/propellants of prolonged storage.

Page 56.

The additional ones include the working medium/propellants, which ensure the work of turbine, tank pressurization, the scavenging of chamber/camera and conduits/manifolds, opening and closing valves and another work.

Furthermore, working medium/propellants are subdivided as follows.

1. Working medium/propellants whose chemical energy is used in RD, i.e., component of chemical fuel/propellant. They are examined in § 1.2 and 1.6.

2. Working medium/propellants whose chemical energy is not used in RD (Fig. 3.2). They are not the source of heat. Therefore their selection produce only on the basis of the condition the most complete conversions of the heat, applied to the working medium/propellant from without, into the kinetic jet energy. Table 3.1 gives the properties of such working medium/propellants (inert gases, alkali metals, etc.).

As working medium/propellant of RD can serve also high-pressure compressed gas, preliminarily charged/filled into the engine chamber and which escapes in the process of its work behind the nozzle.

Due to the initial state of aggregation working medium/propellants of RD can be solid, liquid and gaseous.

On the temperature range the retention/preservations/maintaining liquid state liquid working medium/propellants subdivide into those high-boiling and being low-boiling.

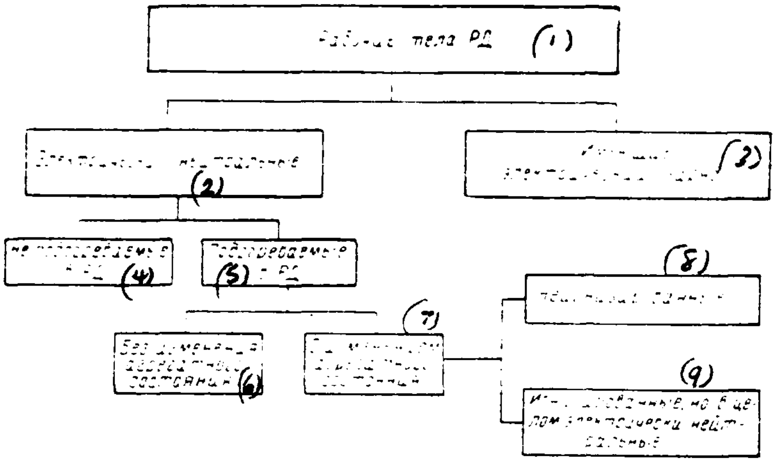

Fig. 3.2. The classification of the working medium/propellants whose chemical energy is not used in RD.

Key: (1). Working medium/propellants in RD. (2). Electrically neutral. (3). Having electric charge. (4). Not preheated in RD. (5). Preheated in RD. (6). Without change in state of aggregation. (7). With change in state of aggregation. (8). Not ionized. (9). Ionized, but as a whole electrically neutral.

Page 57.

Table 3.1. Some properties of working medium/propellants [35], [37].

(1) Рабочее тело	(2) Химическая формула	(3) Порядковый номер элемента	(4) μ, кг/кмоль	ρ, (7) кг/м³: (5) в жидком состоянии	ρ, (7) кг/м³: (6) в твердом состоянии	$T_{пл}$, при нормальном давлении, К (8)	$T_{кип}$, при нормальном давлении, К (8)	(9) Потенциал ионизации атома эв[1]
Гелий (10)	He	2	4,003	470 (при 2°,35 К) (11)	—	0,95 (при давлении 25,5 бар) (12)	4,25	24,58
Неон (13)	Ne	10	20,179	1204 (при 27°,25 К) (11)	—	24,48	27,05; 27,25	21,559
Аргон (14)	Ar	18	39,948	1400 (при 87°,15 К) (11)	1650 (при 40°,15 К) (11)	83,75; 83,85	87,25	15,755
Криптон (15)	Kr	36	83,800	2155 (при 120°,25 К) (11)	—	115,95; 116,05	119,75; 119,95	13,996
Ксенон (16)	Xe	54	131,300	3520 (при 164°,15 К) (11)	—	161,35	165,05	12,127
Азот (17)	N_2	7	28,013	808 (при 77°,15 К) (11)	—	63,15	77,35	14,54
Литий (18)	Li	3	6,939	—	534 (при 298°,15 К) (11)	452,15; 453,15	1623,15; 1643,15	5,39
Натрий (19)	Na	11	22,990	—	972,5 (при 273°,15 К) (11)	370,65; 370,95	1150,15; 1173,15	5,138
Калий (20)	K	19	39,102		860; 850 (при 335°,15 К) (11)	336,75	1038,15; 1049,15	4,339
Рубидий (21)	Rb	37	85,470	1475 (при 311°,95К) (11)	1532	311,95	978,15	4,176
Цезий (22)	Cs	55	132,905	—	1873; 1900	301,65	961,15	3,893
Ртуть (23)	Hg	80	200,590	13546 (11)	—	234,28	629,73	10,434
Вода (24)	H_2O	—	18,016	1000 (при 277°,15 К)	—	273,15	373,15	—

Key: (1). Is working body. (2). Chemical formula. (3). Order number of element/cell. (4). kg/kmole. (5). in the liquid state. (6). in solid state. (7). kg/m³. (8). at normal pressure. (9). ionization potential of atom eating [1].

FOOTNOTE [1]. eV - electron volt; 1 eV = $1.6021 \cdot 10^{-19}$ joule (3.8276•10 20

cal.). ENDFOOTNOTE.

(10). Helium. (11). with. (12). (at pressure of 25.5 bars). (13). Neon. (14). Argon. (15). Krypton. (16). Xenon. (17). Nitrogen. (18). Lithium. (19). Sodium. (20). Potassium. (21). Rubidium. (22). Cesium. (23). Mercury. (24). Water.

Page 58.

To high-boiling relate the working medium/propellants, which under operating conditions on the earth/ground (at the normal atmospheric pressure) have a boiling point of higher than 298°K (25°C), i.e., they are under normal conditions liquid [23].

Low-boiling are called the working medium/propellants which under standard conditions have a boiling point of lower than 298°K (25°C) and are gas. However, some of them can be employed as those high-boiling during the maintenance of a comparatively small pressure in the tank (for example, ammonia NH_3) or in the case of the high freezing point of working medium/propellant (for example, nitrogen tetroxide N_2O_4) - by preheating the tank.

Low-boiling working medium with boiling point below 173°K are separated into the group of so-called cryogenic working

medium/propellants. They include the lowered gases: oxygen, hydrogen, fluorine, helium, nitrogen, etc.

The high-boiling working medium/propellants, physically and chemically stable during the prolonged (of up to several years) shelf-life, calls worker by the bodies of prolonged storage, remaining (and first of all cryogenic) - by working medium/propellants of short-time storage.

§ 3.2. General requirements for the working medium/propellants.

The general requirements, presented to all working medium/propellants, utilized in RD, include:

1. High density. The greater the density of working medium/propellant, the less the volume of capacity or chamber/camera for its arrangement/position. With an increase in the density of working medium/propellant is decreased the mass of tank compartment, which raises the characteristic velocity of rocket vehicle.

2. Physical stability (stability). Working body is physically stable, if in the range of ambient temperatures, in which is employed RD in the composition of rocket vehicle, is provided its necessary state of aggregation. The usually indicated range encompasses

temperature from 213°K [-60°C] to 333°K [+60°C]. For example, solid fuel RDTT must not be softened at elevated temperatures of the environment, but liquid propellant components must retain their homogeneity (not to be exfoliated), not give solid precipitation and not to vaporize.

Liquid working medium/propellants are physically stable at a low temperature of their freezing (or, that one and the same, melting) and to high boiling point.

Page 59.

The use/application of working medium/propellants, which do not possess physical stability under conditions RD raises in price construction/design and operation; for example, in the case of applying the low-boiling (especially cryogenic) working medium/propellants they are necessary:

a) the heat insulation of tanks and mains of RD, which increases the dry mass of rocket vehicle;

b) the system of cooling for decreasing the losses of working medium/propellant to the vaporization prior to the start of rocket vehicle, or the system of the topping of tanks for the

completion/replenishment of the losses indicated;

c) the use/application of structural materials, which possess sufficient impact toughness at low temperatures.

Requirement for the physical stability of working medium/propellants is essential in essence for the combat missiles. For the carrier rockets of KA and for KA themselves the use/application of cryogenic working medium/propellants in view of their high effectiveness from other parameters is not only justified, but in the majority of the cases by optimal version.

3. Chemical stability. In the working medium/propellant must not occur the chemical reactions, which lead to the liberation from its composition of other products.

4. Simplicity of storage, transport and operation. Working body satisfies this requirement if:

a) the vapors of working medium/propellant in the mixture with the air are nonexplosive;

b) is working body and its vapors are nontoxic (they are nontoxic) and do not act on skin, eyes, etc., i.e., they are harmless

for the service personnel;

c) working body does not explode under the influence on it of impact load, it is low-sensitivity to the pollution/contamination and it is not aggressive with respect to the structural materials.

Furthermore, when selecting of working medium/propellant it is necessary to consider its cost/value and mastery by domestic manufacture.

The working medium/propellants, used for the engines of KA, must possess low sensitivity to the cosmic radiation.

§ 3.3. Specific requirements for the working medium/propellants.

Requirements for the working medium/propellants depend also on the type of the engine in which they are used.

The fuels/propellants, utilized in chemical RD, must provide a large quantity of heat, isolated with the course of the reaction of the reaction 1 kg. of propellant components, and the low molecular weight of reaction products, moreover it is desirable, that they during the motion along the nozzle ware in gaseous state. The fuel/propellant, which satisfies these requirements, provides the

high value of specific jet firing.

Working medium/propellants nonchemical thermal RD must possess in exhaust jet analogous with chemical RD low molecular weight, and also low values of specific heats of phase transformations (melting and vaporization).

The working medium/propellants, which satisfy these requirements, provide the high values of specific jet firing, moreover for its work proves to be sufficient the presence onboard the LA of a source of thermal energy of a comparatively low power and, therefore, relatively small mass.

Page 60.

The first requirement is important for all types thermal RD, the second for some engines does not have special importance. For example, nuclear fission of atoms in the reactor YaRD with their low general/common/total mass gives a significant quantity of heat, and problem consists only of working body receiving and was abstracted/removed heat from the zone of reaction and thereby it provided the normal operating temperature of reactor.

When selecting of working medium/propellant for the engines,

which use its electrical properties, are considered the special features/peculiarities of the ionization of working medium/propellant and subsequent dispersal/acceleration of its electrically charged particles. For such working medium/propellants they are necessary:

1) least possible energy content, necessary for the ionization (for the electron detachment from external atom shell), i.e., possibly lower ionization potential;

2) high electroconductivity in the plasma state;

3) the relatively larger mass of electrically charged particles so forth.

Are very specific requirements for the working medium/propellants of the photon engine: the products of their reaction must possess the ability of intensive radiation/emission, i.e., initial energy of working medium/propellants must most completely be converted into the radiant energy.

Page 61.

Part II.

THEORY OF THERMAL ROCKET ENGINES.

Chapter IV.

Real thermodynamic processes of expanding the chemically active (reacting) gases.

§ 4.1. Reactions in the chemically active gases.

The processes of expansion occur differently, depending on temperature and composition of gas or mixture of gases.

During the expansion of cold simple gas (for example, hydrogen) in it it does not occur any chemical reactions, i.e., the composition of gas in different states does not change. This gas call chemically inert. However, if we raise the temperature of this gas, then in it begin to flow/occur/last chemical reactions, which it is possible to

judge, after conducting spectral analysis. For example, in the case of heating hydrogen to high (6000°K) temperature in YaRD and thermal type ERD in its composition appears, besides molecular hydrogen H_2, also atomic hydrogen H.

Chemical reaction products in the chambers/cameras chemical RD are the mixture of different gases, heated to 3000-4500°K. At this high temperature in the mixture or gases occur chemical the reactions, which also leads to a change in its composition.

Gases either the mixtures of gases, in which occur chemical reactions, call chemically active, or reacting. In the process of expanding such gases the reserve of their chemical energy is decreased as a result of its partial conversion into the heat.

Let us examine as the example to chemically active mixture the fuel combustion products in the chamber/camera chemical RD. Into their composition enter the products not only of complete (H_2O, CO_2, HF, etc.), but also incomplete combustion (CO, etc.), or oxidative and combustible elements/cells in the molecular and atomic form (O_2, H_2, O, H, etc.). It should be noted that even with the excess of oxygen are revealed the products of incomplete combustion and unreacting combustible elements/cells.

Page 62.

The above is explained by the reversibility of chemical reactions.

Any chemical reaction between the gases occurs both in the straight line and in the opposite direction. For example, the straight/direct and reverse/inverse reaction of the reaction of oxygen and hydrogen:

$$H_2+\frac{1}{2}O_2\rightarrow H_2O;\quad H_2O\rightarrow H_2+\frac{1}{2}O_2$$

or $H_2+\frac{1}{2}O_2\rightleftarrows H_2O$.

The composition of gaseous mixture depends on the speed of occurence of the straight/direct and reverse/inverse reactions: if forward reaction has high speed, then the concentration of water vapor in the reaction gases in the course of time increases/grows, and at the greater rate of reverse reaction it is increased concentration of H_2 and O_2.

For further calculations is of interest the examination of this state of the mixture of gases, in which the rates of straight/direct and reverse/inverse reactions are equal to each other. This state is called chemical equilibrium.

The chemically equilibrium mixture of gaseous products is characterized by the fact that, in spite of the course of straight/direct and reverse/inverse chemical reactions, its composition at constants temperature and pressure does not change. Into this mixture enter not only the products of the complete or incomplete reaction of initial working medium/propellants, but also the elements/cells of initial working medium/propellants in the molecular or atomic form.

The reactions as a result of which from the products of the complete reaction of working medium/propellants are formed the elements/cells of initial working medium/propellants in the molecular form or from the elements/cells of initial working medium/propellants in molecular form - atomic substances, call the reactions of dissociation. They flow/occur/last with the ingress of heat.

Besides chemical reactions, in the gaseous reaction products of working medium/propellants can flow/occur/last phase reactions and reactions of ionization.

Phase are called the reactions in course of which changes the state of aggregation of reaction products, namely is formed the condensed (liquid or solid) phase. This process is accompanied by the liberation of heat.

The condensed substance in contrast to the gas virtually does not change its volume with a change in the temperature. Therefore it cannot accomplish work of expansion, but accepts participation in the physicochemical processes, which take place in the reaction products. The condensed substance always takes up by vaporization or accumulates by condensation the same substance, which is found in the vaporous state and which participates in the reactions.

Page 63.

Therefore, if process with the phase reactions continues at an invariable temperature, the partial pressures of substances in the mixture of gases always are maintained by invariable ones and equal to the pressure of the saturated steams.

The reactions of ionization appear during heating of gaseous products to the high temperature (for example, in the chamber/camera YaRD and ERD).

As a result of reacting the ionization of the atoms of any chemical element, which takes the form

$$Э \rightarrow Э^{+} + e^{-} + E_{\text{ион}},$$

where $Э^+$ - positively charged/loaded ion;

e^- - electron;

$E_{ион}$ - ionization energy, are formed ions and electrons (electron gas). These reactions occur with the ingress of heat.

§ 4.2. Determination of the composition of chemically active gases.

According to law of mass action the chemical reaction rate is directly proportional to the concentrations of the initial reactants, each of which is taken to the degree, equal to the stoichiometric coefficient with which the substance enters into the equation of chemical reaction.

The content of separate gases in the composition of gaseous products is expressed by their concentration or partial pressures. The equation of chemical reaction it is possible to write in general form

$$a\text{M} + б\text{H} \rightleftarrows в\text{Л} + г\text{C},$$

where M and H - the parent substances of reaction;

L and S - the end products of reaction.

a, b, c and d - stoichiometric coefficients.

According to law of mass action the rate of straight line and reverse/inverse reaction can be written through the partial pressures of the gaseous substances:

$$U_{пр}=K_1 p_M^a p_H^б; \quad U_{обр}=K_2 p_Л^в p_C^г.$$

where K_1 and K_2 - proportionality factors; them they call the rate constants of straight/direct and reverse/inverse reactions;

p_M and p_H - partial pressures of the parent substances of reaction;

$p_Л$ and p_C - partial pressures of the end products of reaction;

Constants K_1 and K_2 of different reactions have different values and increase/grow with an increase in temperature. Consequently chemical reaction rate also grows by increase of temperature. For chemical of the equilibrium mixture of the gaseous substances

$$U_{пр}=U_{обр}.$$

Page 64.

Hence $$K_1 p_M^a p_H^б = K_2 p_Л^в p_C^г$$

or $$\frac{p_M^a\, p_H^б}{p_Л^в\, p_C^г}=\frac{K_2}{K_1}=K=f(T). \qquad (4.1)$$

Ratio K_2/K_1 is called the equilibrium constant of the reaction, expressed through the partial ones of the pressure of the gaseous substances of reaction.

Equilibrium constants analogous with the reaction rate constants are determined by a type of reaction and by a temperature, at which the reaction occurs.

It is most convenient for the thermodynamic calculations (especially for calculations in electronic computers) to use equilibrium constants of the reactions of the dissociation two- and polyatomic gases to atomic; these reactions call the reactions of atomization.

In the overwhelming majority chemical RD are used the fuels/propellants in which are included the following elements/cells: oxygen O, hydrogen H, carbon C, nitrogen N; is promising the use/application of fluorine F. The equations of the atomization of the combustion products of their take the following form:

$$\left.\begin{array}{ll} O_2 \rightarrow 2O; & HF \rightarrow H+F; \\ H_2 \rightarrow 2H; & N_2 \rightarrow 2N; \\ OH \rightarrow O+H; & NO \rightarrow N+O; \\ H_2O \rightarrow 2H+O; & CO \rightarrow C+O; \\ F_2 \rightarrow 2F; & CO_2 \rightarrow C+2O. \end{array}\right\} \quad (4.2)$$

Values of the equilibrium constants of the reactions are introduced in handbook [15].

§ 4.3. Effect of temperature and pressure on the dissociation of the mixture of gases.

The composition of the chemically active mixture of gaseous products thermal RD is determined not only by the composition of working medium/propellant (for chemical RD - by composition of propellant components and by their relationship/ratio), but also by the conditions of the course of the reactions of dissociation.

With a temperature rise of gaseous products their dissociation occurs more intensive. This must be considered during the thermal design thermal RD, in particular during the determination of the temperature in the chamber/camera chemical RD.

With an increase in the temperature in the chamber/camera of heat engines first of all dissociate such products, as H_2O and CO_2

with formation OH, CO H_2, O_2. With further temperature rise dissociate the molecular gases H_2, O_2 and N_2 with the formation of atomic gases.

Page 65.

As a result of the ingrass of heat the dissociation in the chamber/camera chemical RD dacreases the total quantity of heat which is isolated in the presence of the chemical reactions. For example, the combustion products of any fuel/propellant, which consists of elements C, H, O and N, begin noticeably to dissociate with T>2500°K, moreover heat is atsorbed so intensively, that the temperature of the products of combustion of such fuels/propellants cannot exceed 3600°K (in the absence of dissociation it could be considerably above). However, together with a reduction/descent in the temperature of gaseous products dissociation causes certain positive effect/action, namely: it decreases the apparent molecular weight of mixture, since the molecular weight of the formed gaseous products is lower than the mass of the initial products of the reaction of dissociation. However, in whole reaction of dissociation impair the characteristics chemical RD.

In order to decrease (to suppress) the dissociation, select elevated pressures p_K (p_K>100 tar [~~100 kg/cm²]) and such component

of fuel/propellant the reaction products of which they have a comparatively low temperature or struts with respect to the dissociation. To the latter it relates, in particular, hydrogen fluoride HF - reaction product of the reaction of fluorine and hydrogen.

In contrast to chambers/cameras chemical RD dissociation in the chamber/camera YaRD, in which is isolated a significant quantity of heat, it is the positive factor, which ensures the storage of greater energy of working medium/propellant at its limited temperature as a result of the increase of chemical energy. The subsequent partial conversion of this energy in the process of expansion into the heat leads to the appropriate increase in the kinetic energy of working medium/propellant and specific jet firing.

§ 4.4. Equilibrium process of expanding the mixture of gases.

In the process of expanding the chemically active mixture of gases in the nozzle of chamber/camera thermal RD the temperature of mixture descends. Since each is gas mixture tends toward state of chemical equilibrium, then in the mixture of gaseous products during its motion along the nozzle occur the so-called reactions of recombination, which are actually the reactions of burning.

If during the dissociation the part of the heat of gaseous products is converted into the chemical energy, then as a result of reacting the recombination, on the contrary, the chemical energy of gaseous products partially is converted into the heat. The reactions of the recombination of gaseous products compensate to a certain extent of the expenditure of the heat which are caused by their dissociation prior to the nozzle entry. The incomplete reimbursement of the expenditures indicated is explained by the fact that the heat as a result of reacting the recombination during the motion of gaseous products along the nozzle is supplied to them at a pressure less than p_K.

page 66.

As a result of the recombination of gaseous products during the motion nozzle their composition continuously changes. In this case are possible the following cases.

1. In process of expanding gaseous products, during which their temperature and pressure are decreased, composition of products changes accurately in accordance with conditions of chemical equilibrium. This process of expansion they call equilibrium. It has most important value in the theory thermal RD, since in most of the cases the real process of expansion in the nozzle of chamber/camera

is most close precisely to the equilibrium.

Equilibrium expansion is energetically most advantageous, since as a result of the reactions of recombination during this expansion most completely is used the chemical energy of gaseous products.

The equilibrium process or expansion is actually expansion with the delivery of heat to the gaseous products. the curve of process is arranged/located in coordinates p-v between the curves n=k (adiabatic process) and n=1 (isobaric process)

$$1<n_p<k,$$

where n_p - index of the equilibrium process of expansion.

The equation of the equilibrium process of expansion has the form of

$$pv^{n_p}=\text{const.} \qquad (4.3)$$

2. In process of expanding reaction gases recombination does not flow/occur/last. This process is called maximally unbalanced. The composition of gaseous products in this case does not change: the partial pressures of gaseous products are decreased, but their relationships/ratios remain the same as at the nozzle entry, although the temperature of gaseous products during their motion along the nozzle continuously is depressed.

The maximally unbalanced process of expansion in the absence of heat exchange with the chamber walls is completely analogous adiabatic: the chemically active mixture of gaseous products behaves in the maximally unbalanced process of expansion as chemically inert working body.

As can be seen from Fig. 4.1, work of expansion for the process with index k in one and the same interval of pressures is less than for the process with the index n_p.

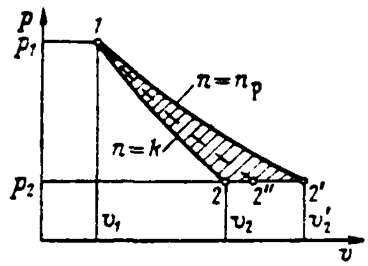

Fig. 4.1. Curves of the processes of the maximally unbalanced (1-2) and maximally equilibrium to (1-2') expansion of gaseous products.

Page 67.

During the maximally unbalanced expansion gaseous products which escape behind the nozzle, take away with themselves the entire chemical energy which during the equilibrium expansion is converted into the heat. Therefore this expansion is least profitable from the point of view of the energy engine characteristics.

3. In process of expanding reaction gases recombination flows/occurs/lasts only partially. This expansion is called partially unbalanced; its curve is arranged/located in coordinates p-v between curves of equilibrium and maximally unbalanced of the processes of expansion. It is depicted in Fig. 4.1 as dotted line 1-2".

§ 4.5. Fundamental equations of the equilibrium process of expanding

the mixture of gases.

The chemically active mixture, the being mixture, which is the mixture of perfect gases, is subject to the equation of state of the perfect gas

$$\frac{p_{\Sigma}}{\rho}=R_{\Sigma}T, \qquad (4.4)$$

where p_{Σ}, ρ, R_{Σ} and T - pressure, density, gas constant and temperature of the mixture of gases.

Gas constant R_{Σ} is connected with the apparent molecular mass μ_{Σ} with the following by relationship:

$$R_{\Sigma}=\frac{8315}{\mu_{\Sigma}}. \qquad (4.5)$$

Value μ_{Σ} can be determined, if are known pressure the mixtures of gases p_{Σ}, partial pressure p_i and molecular weight μ_i of each of them, according to the following equation:

$$\mu_{\Sigma}=\frac{1}{p_{\Sigma}}\sum_{i=1}^{i=n}\mu_i p_i. \qquad (4.6)$$

During the motion of the chemically active mixture of gases along the nozzle its composition changes, which leads to a change in values R_{Σ} and μ_{Σ}, i.e. in contrast to the chemically inert mixture of gases $R_{\Sigma}\neq\mathrm{const}$ and $\mu_{\Sigma}\neq\mathrm{const}$.

For the equilibrium process of expansion value R_{c} it is possible to determine from two parameters of state (for example, T and p); however, this determination in practice proves to be sufficiently to difficult ones.

For the unbalanced process of expanding the mixture of gases must be prescribed/assigned the law, which links the value R_{c} and other parameters of the mixture of gases.

Page 68.

Index n_{p} determine from the known parameters of the mixture of gases at the nozzle entry and at the output/yield from it. We use equations (4.3) and (4.4) and let us fulfill some algebraic conversions, as a result we will obtain

$$n_{\text{p}}=\frac{\lg\frac{p_{\text{к}}}{p_{\text{c}}}}{\lg\frac{R_{\text{c}}T_{\text{c}}p_{\text{к}}}{R_{\text{к}}T_{\text{к}}p_{\text{c}}}} \qquad (4.7)$$

or taking into account relationship/ratio (4.5)

$$n_{\text{p}}=\frac{\lg\frac{p_{\text{к}}}{p_{\text{c}}}}{\lg\frac{\mu_{\text{к}}T_{\text{c}}p_{\text{к}}}{\mu_{\text{c}}T_{\text{к}}p_{\text{c}}}}. \qquad (4.8)$$

Let us write the equation of the mass flow rate of the gas through arbitrary sector of nozzle, which has area f:

$$\dot{m}=f\varrho W. \qquad (4.9)$$

Let us introduce the following designations:

ε_c - expansion ratio of gas in the nozzle;

$\varepsilon_{кр}$ - critical pressure differential;

$\bar{f}_c$ - nozzle expansion ratio.

For calculating the parameters indicated are used the following equations:

$$\varepsilon_c = \frac{p_к}{p_c}; \tag{4.10}$$

$$\varepsilon_{кр} = \frac{1}{\left(\frac{2}{n_р+1}\right)^{n_р/(n_р-1)}}; \tag{4.11}$$

$$\bar{f}_c = \frac{f_c}{f_{кр}}. \tag{4.12}$$

From gas dynamics it is known that if is provided condition $p_к/p_{кр} < \varepsilon_{кр}$, then in what cross section of nozzle it is not possible to drive away gas to the speed, equal to the local velocity of sound a, i.e., in any cross section W<a; if nozzle has inswept and divergent sections, then the gas velocity during the motion in the latter is decreased, and its pressure increases/grows, moreover nozzle does not create thrust.

under condition $p_к/p_{кр}=\varepsilon_{кр}$ the pattern of the flow of gas in the expanding section of nozzle sharply changes the gas velocity continues to be increased, and its pressure, the temperature and density - to be decreased. in this case the nozzle provides supersonic discharge velocity ($W>a$, where a - speed of sound in nozzle exit section) creates thrust.

Page 69.

Under condition $p_к/p_{кр}=\varepsilon_{кр}$ the density of gas ρ continuously falls, and its speed also continuously grows, but value ρW in the inswept and expanding sections of nozzle changes differently. In the range of subsonic speeds (from $W_к=0$ when $f_к\to\infty$ to $W_{кр}=a_{кр}$ when $f=f_{кр}$) product ρW increases/grows with an increase in speed W; in critical section it is maximal. In the range of supersonic speeds (from $W_{кр}$ to W_a) the product ρW with an increase in the velocity is decreased.

It is thawed as on the basis of equation (4.9)

$$f=\frac{\dot m}{\rho W},$$

moreover value $\dot m$ is constant for each cross section of nozzle, the nozzle must consist of series-connected the inswept and divergent

sections. This nozzle is called Laval nozzle (on im. his inventor - Swedish engineer).

Necessary nozzle expansion ratio $\bar{f}_c$ depends only on index n_p and expansion ratio of gas in nozzle ε_c:

$$\bar{f}_c=\frac{\left(\frac{2}{n_p+1}\right)^{1/(n_p-1)}\sqrt{\frac{n_p-1}{n_p+1}}}{\sqrt{\frac{1}{\varepsilon_c^{2/n_p}}-\frac{1}{\varepsilon_c^{(n_p+1)/n_p}}}}. \qquad (4.13)$$

In proportion to an increase in necessary value ε_c value $\bar{f}_c$ it is necessary to increase. At one and the same value ε_c necessary value $\bar{f}_c$ the less, the greater the index n_p. Therefore the utilization of working medium/propellants, which ensure large index n_p, makes it possible to decrease the sizes/dimensions and the mass of nozzle.

Values $\bar{f}_c=f(\varepsilon_c, n_p)$ are given in appendix 3.

The nozzle throat area can be calculated from the formula

$$f_{\kappa p}=\frac{\dot{m}\beta}{p_\kappa}, \qquad (4.14)$$

where β - specific pressure impulse, expenditure complex or the complex β, determined according to the equation

$$\beta=\frac{\sqrt{R_\kappa T_\kappa}}{A_{n_p}}. \qquad (4.15)$$

Value A_{n_p} depends only on index n_p, it are found by the formula

$$A_{n_p} = \sqrt{n_p}\left(\frac{2}{n_p+1}\right)^{(n_p+1)/2(n_p-1)}. \qquad (4.16)$$

Page 70.

Complex β it is possible to calculate from equation (4.15) or to determine experimentally (according to the results of engine testing), using formula (4.14):

$$\beta = \frac{p_к f_{кр}}{\dot{m}}. \qquad (4.17)$$

A comparison of the real and calculated (ideal) values of complex β can be used for the evaluation of the perfection of the processes, which take place in the section of chamber/camera before the critical cross section (see § 4.7).

Complex β for chemical RD in essence depends on composition of fuel/propellant; for the bipropellant it is determined not only by the type of components, but also by the coefficient of their mass relationship/ratio $\varkappa$:

$$\varkappa = \frac{\dot{m}_{ок}}{\dot{m}_т}, \qquad (4.18)$$

where $\dot{m}_{ок}$ and $\dot{m}_т$ - mass oxidizer consumption and fuel per second respectively.

Complex β can serve to a considerable degree as the thermodynamic characteristic of chemical fuel/propellant.

On the basis of equation (4.14) it is possible to make following conclusions.

1. Necessary area $f_{кр}$ increases with increase in flow of gas (or, that one and the same of expenditure of initial working medium/propellant), of complex β and with decrease of necessary pressure $p_к$.

2. For increasing pressure $p_к$ necessary to increase expenditure of $\dot{m}$ or to decrease area $f_{кр}$.

3. Expenditure $\dot{m}$ and pressure $p_к$ vary directly (if we disregard/neglect certain dependence of complex β from pressure $p_к$).

One should emphasize that in accordance with equations (4.13) and (4.14) the expansion ratio of gas in the nozzle does not depend on its expenditure: with an increase in the expenditure simultaneously increase/grow values $p_к$ and p_c (and vice versa), but relation $p_к/p_c$ remains invariable.

The equation of Bernoulli for 1 kg. of gas, based on the law of

conservation of energy, in connection with cross section at the nozzle entry (we accept $W_{к}=0$) and to its exit section has the following form:

$$i_{п\,к}=i_{п.с}+\frac{W_c^2}{2}, \qquad (4.19)$$

where $i_п$ - the total enthalpy of gas, which is generalized, parameter, which includes enthalpy, and chemical energy of working medium/propellant.

Page 71.

The total enthalpy of the chemically active mixture of gases can be calculated from the equation

$$i_п=c_p'T, \qquad (4.20)$$

where c_p' - specific heat of the mixture of gases for the isobaric process taking into account the course of the reactions of dissociation.

Gas enthalpy i is called the value, equal to the sum of its internal energy u and potential pressure energy pv (where v - the specific volume of gas), i.e.,

$$i=u+pv.$$

Gas enthalpy can be expressed also by the equation

$$i=c_pT,$$

where c_p - specific heat of gas for the isobaric process without taking into account the course of the reactions of dissociation, i.e., for the chemically inert gas.

The relation of specific heat capacity c_p and specific heat for isochoric process c_v is equal to the index of the adiabatic process k:

$$k=\frac{c_p}{c_v}.$$

At condition $W_{\kappa}=0$ all other parameters of gas have maximally possible for this flow values, i.e., are stagnation parameters (T^*, p^* and so forth). In many instances speed W_c is considerable general speed W_{κ} so that the latter can be disregarded/neglected (i.e. to count $W_{\kappa}=0$) without significant damage for the precision/accuracy of calculation. Subsequently under the parameters with the index "to" we will understand stagnation parameters, but lower for simplicity of presentation sign *.

The exhaust gas velocity behind the nozzle can be determined from equation (4.19):

$$W_c=\sqrt{2(i_{\text{п.к}}-i_{\text{п.с}})}. \qquad (4.21)$$

Formula for calculating the exhaust gas velocity behind the nozzle can be obtained also from the equation of Bernoulli, written in the following form:

$$\frac{n_p}{n_p-1} R_к T_к = \frac{W_c^2}{2} + \frac{n_p}{n_p-1} R_c T_c.$$

Hence

$$W_c = \sqrt{\frac{2n_p}{n_p-1} R_к T_к \left(1 - \frac{R_c T_c}{R_к T_к}\right)}.$$

Page 72.

On the basis of equations (4.3), (4.4) and (4.10) it is possible to write the following relationship/ratio:

$$\frac{R_c T_c}{R_к T_к} = \frac{1}{\varepsilon_c^{(n_p-1)/n_p}}. \qquad (4.22)$$

Therefore

$$W_c = \sqrt{\frac{2n_p}{n_p-1} R_к T_к \left(1 - \frac{1}{\varepsilon_c^{(n_p-1)/n_p}}\right)}. \qquad (4.23)$$

The analysis of equation (4.23) shows that speed W_c is increased with an increase in values $R_к$, $T_к$, ε_c and decrease in index n_p. But an increase in temperature $T_к$ leads to the complication of cooling chamber/camera, and with the increase of the expansion ratio of gas in the nozzle and with the decrease of index n_p are increased sizes/dimensions and mass of nozzle. Therefore speed W_c it is most expedient to increase via an increase in gas constant $R_к$ (or, that one and the same, by decreasing the apparent molecular weight of the mixture of gases $\mu_к$).

The maximum speed of gas at prescribed/assigned values n_p, R_κ and T_κ can be achieved/reached in the purely theoretical case - in the absence of any losses and when $\varepsilon_c \to \infty$, and the latter is possible only for infinite nozzle $(\bar{f}_c \to \infty)$. In this case on the basis of equation (4.23)

$$W_{c\,max} = \sqrt{\frac{2n_p}{n_p - 1} R_\kappa T_\kappa}. \qquad (4.24)$$

§ 4.6. Energy losses in the thermal rocket engines.

From the thermodynamics it is known that the cycle of any heat engine is called the sequence of the thermodynamic processes, which occur in the working medium/propellant of engine and which lead to the conversion of heat into the work.

Let us examine the cycle of thermal rocket engine (Fig. 4.2) with the following special features/peculiarities.

1. Initial is working body (for chemical RD - propellant components) is liquid and is supplied into barrel.

2. Nozzle exit pressure is equal to ambient pressure $(p_c = p_h)$. This mode/conditions the work of the nozzle of chamber/camera thermal RD

call calculated.

We will examine case $p_c=p_h\neq0$, for which the nozzle, which operates in the design conditions, it has finite dimensions. Initial pressure of working medium/propellant $p_A=p_c=p_h$. Into cycle thermal RD enter four following basic processes.

1. Process AB is process of increasing pressure (compression) of liquid working medium/propellant.

Page 73.

2. Process BK is process of delivery of heat to working medium/propellant when $p_B=p_K=\text{const}$.

3. Process KS is process of equilibrium expansion of gaseous working medium/propellant with index k on pressure p_K up to a pressure of p_c.

4. Process SA can be presented only theoretically, if we consider that via branch/removal of heat from gaseous working medium/propellant, which has parameters p_c, T_c and ϱ_c, it with the aid of special devices is continuously condensed and is cooled to state A. At this case process CA would be evaluated as the process of

compression when $p_c=p_A$=const. in which heat it is abstracted/removed from the working medium/propellant.

Area ABKSA on certain scale numerically is equal to the complete (available) work 1 kg. of working medium/propellant in the cycle of thermal rocket engine: this cycle is theoretical. It does not consider losses, with exception of the loss of the total enthalpy, taken away together with the working medium/propellant, which escape/ensues from the nozzle.

Let us examine the expenditures of energy and loss, specific to the processes, which take place in the thermal rocket engines.

Process of the compression of working medium/propellant. If we plot along the axis v specific volume of liquid working medium/propellant, then area Abba is numerically equal to the work which must be completed for compression and supplying the working medium/propellant into the engine chamber.

Process of the delivery of heat to the working medium/propellant. In the actual engine the heat to the working medium/propellant is supplied not in the process BK, while in the process BK; inherent to it are specific losses, caused by;

a) by the dissociation of gaseous products;

b) by the dispersal/acceleration of gaseous products prior to the nozzle entry, which leads to a drop in the pressure (in accordance with the equation of Bernoulli);

c) by the friction of working medium/propellant against the chamber walls (on section prior to the nozzle entry);

d) by the incompleteness of the course of the chemical reaction of burning or decomposition (for example, in RD, the worker on the bipropellant, in connection with the impossibility of the ideal mixing of components) or the incompleteness of the transfer of heat from its source to the working medium/propellant.

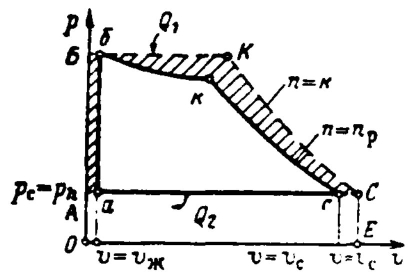

Fig. 4.2. Ideal (ABKS) and real (abKs) cycles thermal RD.

Page 74.

Due to indicated losses the heat to the working medium/propellant thermal RD is supplied not in isobaric process (p=const), but in the process which by coordinates p=v is evinced by curve of ~~ß~~ b K. The losses of enthalpy in the process indicated are determined by area bKk. They cause the decrease of specific pressure impulse β.

Process of expanding the gaseous products. The basic losses of this process are caused:

a) by the carry-off of the total enthalpy together with the working medium/propellant. This loss is unavoidable both with the real and during the ideal cycle. Area CEOA is numerically equal to the unused enthalpy indicated;

b) by the nonparallelism of the streams of the working axle center of the nozzle (see Fig. 1.o);

c) by the friction of working medium/propellant against the nozzle liners;

d) by the effect of the tapering portion of the nozzle (entry loss into the nozzles);

e) by the inequilibrium of the process of expansion;

f) by the branch/removal of heat of the working medium/propellant in wall of chamber/camera (loss to the nonadiabaticity of the process of expansion);

g) by the formation of the condensed phase in the process of moving the working medium/propellant along the nozzle.

The losses indicated cause the decrease of thrust coefficient K_P.

§ 4.7. Efficiencies of chemical rocket engines.

Losses in chemical RD estimate by energy and by pulse efficiencies.

Energy efficiency (efficiency of cycle and others) characterize energy losses in the engine.

The efficiency of cycle $\eta_ц$ is the ratio of the work of cycle $L_ц$ to chemical energy, which is contained in 1 kg. of fuel/propellant. The energy indicated we will call net calorific power. Let us designate it $H_{раб}$. Then

$$\eta_ц = \frac{L_ц}{H_{раб}}. \tag{4.25}$$

The efficiency of cycle it is possible to express also by the formula

$$\eta_ц = \eta_г \eta_с \eta_t, \tag{4.26}$$

where $\eta_г$ - efficiency characterizing losses in the chamber/camera chemical RD, the lowering specific pressure impulse β;

η_t - thermal efficiency considering the losses, connected with the carry-off of the total enthalpy together with the reaction products behind the nozzle;

η_{r} - efficiency characterizing other losses in the nozzle, which lead to the decrease of the thrust coefficient of chamber/camera K_{P}.

Page 75.

Efficiency η_{β} estimates the degree of completeness of the transformation of the chemical energy 1 kg. of fuel/propellant into heat Q:

$$\eta_{\beta}=\frac{Q}{H_{\text{раб}}}. \qquad (4.27)$$

Efficiency η_c and η_t are determined the efficiency of the process of the expansion:

$$\eta_{\text{п.р}}=\eta_c\eta_t. \qquad (4.28)$$

Efficiency $\eta_{\text{п.р}}$ characterizes the degree of completeness of the transformation of the heat, which was isolated as a result of chemical reaction in the chamber/camera, in the works of cycle:

$$\eta_{\text{п.р}}=\frac{L_{\text{ц}}}{Q}. \qquad (4.29)$$

Let us examine thermal efficiency η_t, considering losses due to the carry-off of the total enthalpy by reaction products as a result of the finite dimensions of nozzle. Since to ideal cycle are specific only these losses, then $\eta_t=\eta_{\text{ид}}$, where $\eta_{\text{ид}}$ - efficiency of ideal cycle.

In purely theoretical case ($f_c\to\infty$; $p_c=p_h=0$) are absent all forms of

losses) entire/all heat, which was isolated as a result of chemical reaction, it is converted into the kinetic energy of reaction products (into the kinetic jet energy), i.e., for 1 kg. of the fuel/propellant

$$H_{\text{раб}}=\frac{W^2_{c\,max}}{2} \qquad (4.30)$$

or

$$W_{c\,max}=\sqrt{2H_{\text{раб}}}. \qquad (4.31)$$

Since in the case $p_c \to 0$, in question then in accordance with equations (1.20) and (1.4)

$$I_{\text{уд}\,max}=W_{c\,max}$$

or taking into account relationship/ratio (4.31)

$$I_{\text{уд}\,max}=\sqrt{2H_{\text{раб}}}. \qquad (4.32)$$

The ideal exhaust velocity of the chemically active mixture of gases is called the speed at which consider only the phenomena, connected with the dissociation and the recombination, and the losses, caused by the finite dimensions of nozzle, i.e., the losses, connected with the nose of the total enthalpy together with the gases, which escape behind the nozzle. Let us designate the speed W_{ct}. indicated Kinetic jet energy is equal to

$$W^2_{ct}/2.$$

Page 76.

Thermal efficiency is the relation of kinetic jet energy to the

working heat productivity:

$$\eta_t = \frac{W_{c\,t}^2/2}{H_{раб}},$$

or taking into account equation (4.31)

$$\eta_t = \frac{W_{c\,t}^2}{W_{c\,max}^2}.$$

By the utilization of relationship (4.23) and (4.24) latter/last equation it is possible to reduce to the following form:

$$\eta_t = 1 - \frac{1}{\varepsilon^{(n_p-1)/n_p}}. \qquad (4.33)$$

Consequently, thermal efficiency depends only on the expansion ratio of gas in the nozzle and on index n_p.

Value η_t is equal to one only for purely theoretical case examined above $(f_c \to \infty;\ p_c \to 0;\ \varepsilon_c \to \infty)$.

The formula of the theoretical specific impulse, which corresponds to the ideal exhaust velocity and considering only losses, estimation of efficiency η_t, can be obtained from equation (4.32):

$$I_{уд\,t} = \sqrt{2\eta_t H_{раб}}. \qquad (4.34)$$

The actual specific impulse, i.e., the specific impulse, considering all forms of losses, is designed from the equation, obtained by replacement $H_{раб}$ in equation (4.34) by product $\eta_\beta \eta_c H_{раб}$:

$$I_{уд\,д} = \sqrt{2\eta_t \eta_\beta \eta_c H_{раб}}. \qquad (4.35)$$

Energy efficiency are used only for qualitative evaluation of the processes, which take place in the chamber/camera of chemical rocket engines.

In calculations of chambers/cameras chemical RD to more conveniently use pulse efficiencies. They estimate the losses of directly specific impulse, namely these losses are of greatest interest. Pulse efficiency designate by letter φ and they frequently call simply coefficient φ.

Pulse efficiency are connected with the appropriate efficiency with the relationship/ratio

$$\varphi=\sqrt{\eta}. \tag{4.36}$$

Page 77.

Coefficient φ_β is the relation of specific pressure impulse for the real chamber/camera and the chamber/camera, which works without the losses to the incompleteness of burning, the friction also of other losses

$$\varphi_\beta=\frac{\beta_x}{\beta_{\text{нх}}}. \tag{4.37}$$

The coefficient indicated is called the coefficient of completeness

of specific pressure impulse (by coefficient of completeness of expenditure complex).

If we use for the real chamber/camera and the chamber/camera, which works without the losses, $\dot{m}_{д}=\dot{m}_{ид}$ and $f_{кр.д}=f_{кр.ид}$, then

$$\varphi_{\beta}=\frac{p_{к.д}}{p_{к.ид}}. \qquad (4.38)$$

Consequently, coefficient φ_{β} characterizes the losses, connected with the imperfection of processes in the combustion chamber (or decomposition) chemical RD and leading to the decrease pressures in the real chamber/camera in comparison with the chamber/camera, which works without the losses. Coefficient φ_{β} is called also by the coefficient of completeness of pressure.

The coefficient of nozzle φ_c is the ratio of thrust coefficients in the vacuum of real chamber/camera and chamber/camera, which works without the losses to the friction and other losses in the nozzle

$$\varphi_c=\frac{K_p}{K_{p_{ид}}}. \qquad (4.39)$$

Taking into account of equation (4.34) and (4.36) it is possible to write formula (4.35) for calculating the real specific impulse through the pulse efficiency in the form

$$I_{уд.д}=\varphi_{\beta}\varphi_c I_{уд\,t}. \qquad (4.40)$$

Usually value φ_{β} and φ_c sufficiently close to unity: $\varphi_{\beta}=0.96-0.99$; $\varphi_c=0,96\div0,98$.

Product $\varphi_\beta \varphi_c$ is called the solidity ratio of specific impulse and designate φ_I, i.e.

$$\varphi_I = \varphi_\beta \varphi_c. \quad (4.41)$$

Coefficient φ_I is equal to 0.92-0.97.

Page 78.

Chapter V.

CHARACTERISTIC MODES OF THE WORK OF THERMAL ROCKET ENGINES.

§5.1. Characteristics of thermal rocket engines.

With the work of engine in the composition of flight vehicle usually changes flight altitude h and, consequently, also ambient pressure p_h. Therefore it is necessary to determine and to consider the dependences of thrust and specific jet firing on the height/altitude of flight i.e. functions $P_h=f(h)$ and $I_{уд\,h}=f(h)$.

In chapter II it was shown that the prescribed/assigned parameters of trajectory of LA are provided by a change in the thrust ofRD, for which respectively changes the mass flow rate of working medium/propellant (propellant components) $\dot{m}$. Therefore during

calculations and operation of RD the high value have the dependences of thrust and specific impulse on the consumption $\dot{m}$, i.e., function $P_h=f(\dot{m})$ and $I_{\text{уд}\,h}=f(\dot{m})$.

Discharge characteristic. The expenditure (throttle) characteristic of thermal rocket engines is called the dependence of their thrust on the mass flow rate per second of working medium/propellant (propellant components) at constant values $\bar{f}_c$ and h, and for ZhRD and RDGT, furthermore, when $\varkappa$=const.

Let us write again equation (1.9)

$$P_h=\dot{m}W_c+f_c(p_c-p_h). \qquad (5.1)$$

A change of pressure $p_к$ into the process of work thermal RD leads to some changes in the course of processes in the chamber/camera, for example to a change of dissociating the gaseous products and, consequently, also values $T_к$, $R_к$ and W_c. However, for the majority thermal RD mass flow rate per second of working medium/propellant changes within comparatively small limits. For such RD it is possible to consider that the discharge velocity W_c on depends on $\dot{m}$.

From the remaining parameters, entering equation (5.1), on consumption of $\dot{m}$ depends only pressure p_c. Let us determine this dependence, i.e., function $p_c=f(\dot{m})$. For each this nozzle ($\bar{f}_c$=const)

expansion ratio the phase in nozzle ε_c is constant. In accordance with equations (4.10) and (4.14)

$$p_c = \frac{\dot{m}\beta}{\varepsilon_c f_{\kappa p}} = \frac{\beta}{\varepsilon_c f_{\kappa p}} \dot{m}. \qquad (5.2)$$

Let us substitute expression (5.2) in equation (5.1) and let us fulfill some conversions:

$$P_h = \left(W_c + \frac{\bar{f}_c\beta}{\varepsilon_c}\right)\dot{m} - f_c p_h. \qquad (5.3)$$

Page 79.

All values, entering in bracket do not depend on consumption of $\dot{m}$. Consequently, the equation of discharge characteristic is the equation of straight line (Fig. 5.1) which only for case $p_h=0$ (i.e. for the case of the work of engine in the vacuum) runs through the origin of coordinates, and in all remaining cases transverse axis P_h so that with $\dot{m}=0$ $P_h=-f_c p_h$.

With significant reduction or flow rate $\dot{m}$ thermal RD can work unstably or with the work at the level of sea due to large overexpansion (see pg. 83) proceeds the flow breakaway of gas from the nozzle liners. In the latter case will change actual value $\bar{f}_c$, i.e. it will be destroyed the initial condition of discharge characteristic. Therefore its equation is correct to certain level of a reduction/descent in expenditure/consumption of $\dot{m}$.

The fictitious section of characteristic is depicted in Fig. 5.1 as dotted line. In actuality in this section curve takes the form, depicted in Fig. 5.1 and 5.2 as dot-dash line.

The angle of the slope of characteristic BC is equal to arctg a, where the factor of proportionality a is equal to bracketed expression in equation (5.3). For this fuel/propellant (working medium/propellant) value a depends only on values W_c and f_c. For each this engine parameters W_c, ε_c and β can be considered taking into account the noted above assumptions constants. Therefore its discharge characteristics, plotted for different heights/altitudes, are the family of the parallel lines (see Fig. 5.1). A difference in the thrust for the work in the vacuum and at the height/altitude with the pressure of atmosphere p_h is equal to $f_c p_h$.

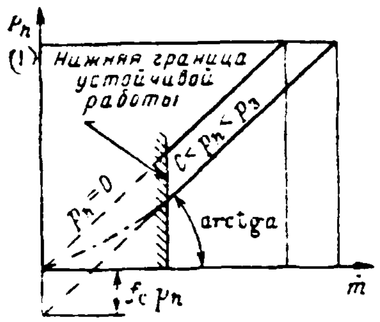

Fig. 5.1.

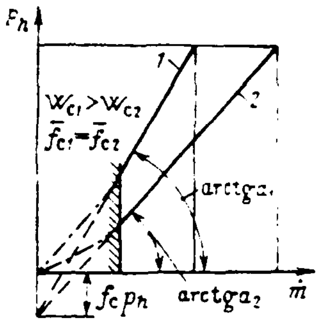

Fig. 5.2.

Fig. 5.1. Discharge characteristics thermal RD with work in vacuum and at height/altitude with pressure p_h

Key: (1). Lower boundary of stable operation.

Fig. 5.2. Discharge characteristics thermal RD $\bar{f}_c=$ const; different values W_c)

Page 80.

On the basis of equation (5.3) thrust in vacuum ($p_h=0$)

$$P_h=\left(W'_c+\frac{\bar{f}_c\beta}{\varepsilon_c}\right)\dot{m}. \qquad (5.4)$$

With one and the same composition of chemical fuel/propellant, and in the case of nonchemical thermal RD - with one and the same

working medium/propellant and quantity of applied heat, the specific impulse can be raised, increasing nozzle expansion ratio $\bar{f}_c$. As it was shown in Chapter IV, with increase $\bar{f}_c$ increases the expansion ratio of gas in nozzle ε_c.

With one and the same sizes/dimensions of nozzle the specific impulse can be increased upon transfer to the fuel/propellant with the larger heating power (chemical RD) or upon transfer to working body with the smaller molecular weight and in the case of the delivery to it of a greater quantity of heat (nonchemical thermal RD).

Fig. 5.2 shows the discharge characteristics of heat engines with the invariable sizes/dimensions of nozzle with the work on fuels/propellants with different heating power. The characteristics indicated proceed from one and the same ficticious point, since in both cases $f_c p_h$=const.

Fig. 5.3 depicts the discharge characteristics of two engines, which are characterized by only nozzle expansion ratio $\bar{f}_c$. These characteristics transverse axis p_h at different points as a result of a difference in values f_c. From Fig. 5.3 it is evident that for one and the same pressure p_h with $\dot{m}<\dot{m}'$ to more advantageous use nozzle 2, and when $\dot{m}>\dot{m}'$ - nozzle 1.

For the construction of discharge characteristic, besides finding of its ficticious point when $P_h=0$, it suffices to find one additional point. It is most convenient to use thrust in the nominal rating, i.e., point $P_h=P_{h\,ном}, \dot{m}=\dot{m}_{ном}$.

Let us examine the dependence of specific impulse thermal RD on the expenditure/consumption of working medium/propellant, i.e., function of the type $I_{уд\,h}=f(\dot{m})$. Let us divide left and the right side of equation (5.3) to m:

$$I_{уд\,h}=W_c+\frac{\overline{f}_c\beta}{\varepsilon_c}-\frac{f_c p_h}{\dot{m}}. \qquad (5.5)$$

This equation takes the following algebraic form:

$$y=a-\frac{b}{x} \text{ или}^{(1)} \ I_{уд\,h}=a-\frac{b}{\dot{m}}.$$

Obtained dependence $I_{уд\,h}=f(\dot{m})$ is the equation of hyperbola (Fig. 5.4).

Value a is equal to specific impulse in the vacuum, i.e., to the maximum specific impulse of this engine; it it is possible to obtain from equation (5.5) for condition $p_h=0$:

$$I_{уд.п}=W_c+\frac{\overline{f}_c\beta}{\varepsilon_c}. \qquad (5.6)$$

Page 81.

Consequently, with the decrease of the expenditure/consumption of working medium/propellant for condition h=const specific jet firing descends, which is explained by deviation from the nominal rating of its work.

Furthermore, with the decrease of expenditure/consumption of $\dot{m}$ and pressure p_K in the chambers/cameras ZhRD is decreased an injector pressure drop, which makes the atomization of propellant components and the process worse of their reaction. However, the caused by the special features/peculiarities indicated reduction/descent in the specific impulse is not considered by equation (5.5).

It is expedient, but is considerably more complicatedly this power change, with which its specific impulse does not descend. For this, for example, ZhRD must provide the following conditions:

1) Δp_Φ=const. i.e., the constant injector pressure drop, which makes it possible not to impair the quality of the atomization of propellant components;

2) p_K=const, i.e., constant pressure p_K, which provides the invariable conditions of the course of the chemical reactions of

reaction in the chamber/camera;

3) $p_c=p_h$; with satisfaction of this condition in the case of the climb specific jet firing increases/grows. This mode/conditions the work of the nozzle of chamber/camera call calculated (see §5.2).

For satisfaction of condition $\Delta p_{\text{ф}}$=const it is necessary to change the discharge area of injectors with respect to a change in the propellant component flow $\dot{m}$. With a reduction/descent in expenditure/consumption of $\dot{m}$ of pressure $p_{\text{к}}$ it is decreased. So that the pressure $p_{\text{к}}$ would remain invariable, it is necessary simultaneously with a reduction/descent in expenditure/consumption of $\dot{m}$ to respectively decrease the nozzle throat area $f_{\text{кр}}$. In turn, for retaining/preserving/maintaining the design conditions of the work of nozzle with decrease $f_{\text{кр}}$ must be proportional reduced the nozzle exit area f_c.

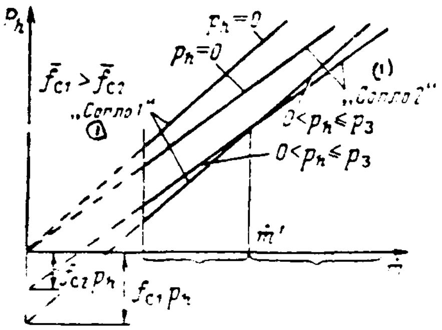

Fig. 5.3. Discharge characteristics thermal RD, which are characterized by only values $\bar{f}_c$

Key: (1). Nozzle.

Page 82.

A power change with the retention/preservation/maintaining of invariable pressure p_K is used very limited due to the large complexity of this change.

If into composition of DU enter several engines, then thrust of DU sufficiently simply can be gradually changed with the disconnection of separate engines during the invariable mode of operation of remaining.

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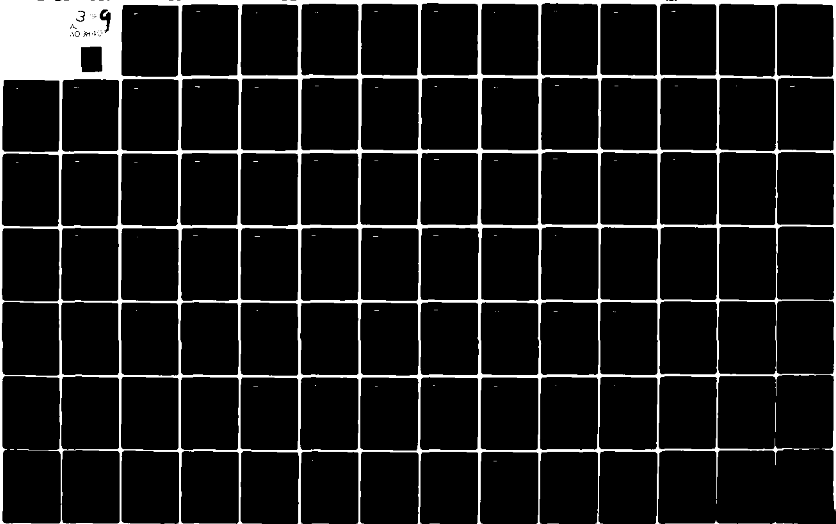
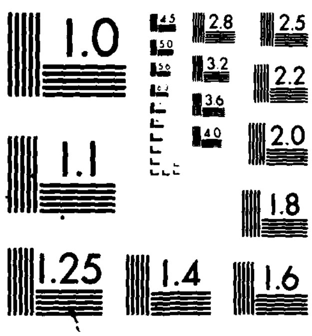



1.0
2.8
2.5
3.2
2.2
3.6
4.0
2.0
1.1
1.8
1.25
1.4
1.6

Pressure $p_{\text{к}}$ in the case of the account of the adopted assumptions directly proportional to the expenditure/consumption of the working medium/propellant

$$p_{\text{к}} \sim \dot{m}.$$

Therefore dependences $P_h = f(p_{\text{к}})$ and $I_{\text{уд}\,h} = f(p_{\text{к}})$ are analogous to dependences $P_h = f(\dot{m})$ and $I_{\text{уд}\,h} = f(\dot{m})$.

Altitude performance. In §1.3 it was shown that the thermal rocket engine, which works in one and the same mode/conditions ($\dot{m}$=const), develops different thrust in the dependence on flight altitude LA.

Dependence thrusts thermal RD on flight altitude $P_h = f(h)$ call altitude performance.

Let us write again equation (1.8)

$$P_h = P_{\text{п}} - f_c p_h. \qquad (5.7)$$

Values $P_{\text{п}}$ and f_c for each given engine are constant. Therefore equation (5.7) is reduced to the equation of the type y=a-bx, moreover $x = p_h$. The pressure of atmosphere p_h to a great degree descends with an increase in altitude. Therefore the engine thrust with an increase in altitude of flight increases/grows.

Altitude performance of thermal rocket engine is depicted in

Fig. 5.5. Specific impulse thermal RD analogous with thrust with an increase in altitude of flight increases/grows, what is their great advantage and it in principle differs thermal RD from all other heat engines.

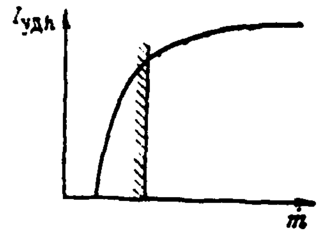


Fig. 5.4.

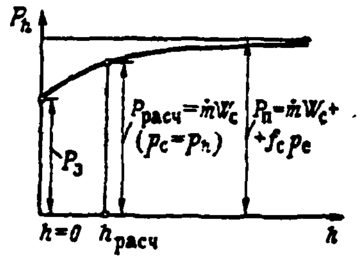


Fig. 5.5.

Fig. 5.4. Dependence of specific impulse thermal RD on expenditure/consumption of working medium/propellant (propellant components).

Fig. 5.5. Altitude performance thermal RD with constant value $\bar{f}_c$.

Page 83.

§5.2. Modes/conditions of the work of the nozzle of chamber/camera.

The nozzle of chamber/camara, which has the prescribed/assigned expansion ratio $\bar{f}_c$, can work under conditions, which are distinguished by the relationship/ratio of the gas pressure in nozzle exit section p_c and ambient pressure at given height/altitude p_h, among other things:

1) $p_c=p_h$. Mode/conditions the work of nozzle under this condition, as has already been indicated, call calculated, and the height/altitude at which the nozzle works in the design conditions, calculated. The pressure of gas p_r determines the so-called nozzle design altitude. The less p_c, the greater the nozzle design altitude;

2) $p_c>p_h$, i.e. gas is expanded in the nozzle up to a pressure of, which is more than ambient pressure (gas expands incompletely). This mode/conditions the work of nozzle call system of insufficient expansion. It is specific to the nozzle, which operates at the heights/altitudes, greater than rated altitude. In particular, under the conditions of underexpansion works nozzle any thermal RD in the vacuum.

System of insufficient expansion can be created, also, with the work of engine on stand. For this it is necessary to raise the thrust of engine whose nozzle works in the design conditions, increasing the expenditure/consumption of working medium/propellant into the chamber/camera. In this case will increase pressure p_K, and, consequently, pressure p_c, so that to be ensured condition $p_c>p_h$;

3) $p_c<p_h$, i.e. gas is expanded in the nozzle up to a pressure of less than ambient pressure. In this case condition $p_c=p_h$ is provided in some intermediate cross section of nozzle. During further motion

along the nozzle at its certain length the gas does not blow away from the walls. Therefore gas is red-enlarge, i.e., in the final part of the nozzle its pressure is less than the ambient pressure. This operating mode occurs at the height smaller than the calculated.

In particular, the gas pressure in nozzle exit section of the engine chambers of first stage of ballistic and space vehicles is usually selected appropriate. Therefore in the beginning of the powered flight trajectory of such rockets the nozzle of chamber/camera and engine works under the conditions of overexpansion.

The system of overexpansion also can be created with the work of engine on stand. For this it is necessary to decrease the thrust of engine whose nozzle works in the design conditions, decreasing the expenditure/consumption of working medium/propellant into the chamber/camera. In this case of pressure $p_к$ and p_c respectively will be lowered and will be ensured condition $p_c<p_h$.

In the process of the climb the engine nozzle of first stage of ballistic missiles first works under the conditions of overexpansion, at rated altitude - some moment/torque of time in the design conditions, and during further climb up to the disconnection - under the conditions of underexpansion, the degree of underexpansion

continuously increasing/growing.

Page 84.

Thrust and specific impulse or each this engine with the invariable nozzle geometry in the nominal rating have great value, if the nozzle of engine chamber works in the design conditions. With the deviation from the design conditions, i.e., under conditions the underexpansion and the overexpansion, value P_h and $I_{уд\,h}$ respectively descend. Therefore when selecting of pressure p_c one should approach that so that the nozzle of engine chamber would work in the design conditions. However, in the majority of the cases this is impossible, and pressure p_c is selected by such so that the losses of specific impulse due to the deviation from the rated nozzle conditions in the powered phase would be smallest.

The value of average/mean specific impulse on the powered flight trajectory can be determined according to the equation

$$I_{уд\,h\,ср} = W_c + \frac{f_c(p_c - p_{h\,ср})}{\dot{m}}, \tag{5.8}$$

where $p_{h\,ср}$ - average/mean on the powered flight trajectory pressure of the atmosphere.

We analyze the modes/conditions of the work of the nozzle of

chamber/camera, using equation (5.1). For the design conditions it takes the form

$$P_h = \dot{m}W_{c\sim}$$

With the work of the nozzle of chamber/camera under the conditions of overexpansion ($p_c < p_h$) product $f_c(p_c - p_h)$ is negative. Therefore the engine thrust with the work of the nozzle of chamber/camera under the conditions of overexpansion is less than in the design conditions. The decrease of thrust indicated can be explained also, examining the diagrams/curves of pressure on the final part of the nozzle (Fig. 5.6). Algebraic composition of forces which effect on the nozzle liner from within and outside, leads to the creation of negative thrust. If we shorten nozzle to lengths 1, i.e., to decrease value f_c, then the engine thrust will increase.

For system of insufficient expansion ($p_c > p_h$) product $f_c(p_c - p_h)$ has positive value. However, in this case thrust is less than in the design conditions. Speed W_c with the work of nozzle under the conditions of underexpansion is so less than in the design conditions, that is provided the inequality

$$\dot{m}W_{c.\text{расч}} > \dot{m}W_c + f_c(p_c - p_h).$$

In order to translate the nozzle of chamber/camera from the system of insufficient expansion to the design conditions at invariable pressure p_k it is necessary to increase expansion ratio f_c, for example by the elongation/aspect ratio of nozzle, which in

accordance with equation (5.3) leads to the increase of thrust. The thrust increment indicated can be explained in the examination of the diagrams/curves of pressure on the imaginary extension of nozzle (Fig. 5.7). Net force with the algebraic addition of the diagrams/curves of the pressure, which effects on the imaginary extension from within and outside, in the direction coincides with the thrust vector. Consequently, the addition of nozzle to the nozzle, which operates under the conditions of underexpansion, makes it possible to increase the engine thrust.

Page 85.

Dependence $I_{уд\,h}=f(p_h)$ can be obtained from equation (5.7) by dividing its right and left side to the expenditure/consumption of the working medium/propellant $\dot{m}$:

$$I_{уд\,h}=I_{уд.п}-\frac{f_c p_h}{\dot{m}}. \qquad (5.9)$$

Consequently, dependences $P_h=f(h)$ and $I_{уд\,h}=f(h)$ are analogous, and if altitude performance is built in the form of graphs $P_h/P_п=f(h)$ and $I_{уд\,h}/I_{уд.п}=f(h)$, then both dependences are depicted as one curve.

Let us additionally explain concept "critical speed" and let us examine the special features/peculiarities of the work of Laval nozzle, for which let us analyze the modes of its operation with different relationships/ratios of the gas pressure in chamber/camera $p_к$

and of ambient pressure p_h.

Let us examine the following conditions for chamber operation:

1) $p_к=p_з$, where $p_з$ - pressure of the atmosphere in the Earth (at the level of sea); pressure $p_к$ is maintained by constant;

2) pressure p_h constantly it is decreased.

Such conditions can be provided, if to build up the chamber/camera indicated into the upper air in the composition of any rocket.

At a pressure of the atmosphere, only somewhat smaller than the pressure $p_к$, of the gas velocity in the nozzle small; they considerably lower than speed of sound, moreover the gas flow is low. In proportion to lowering the pressure of the atmosphere the gas velocity in the nozzle increases/grows, remaining in all cross sections lower than the local velocity of sound. In this case respectively grows the gas flow. Nozzle works as the Venturi tube: the gas velocity in the tapering portion of the nozzle increases/grows, and in that expanding - it is decreased, the nozzle creating no thrust.

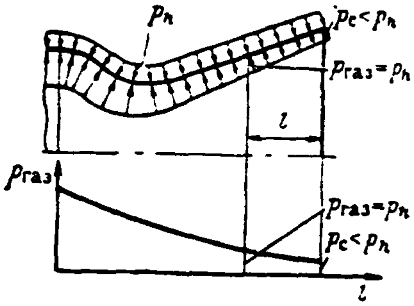


Fig. 5.6.

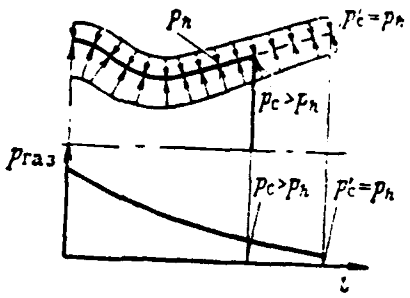


Fig. 5.7.

Fig. 5.6. Diagrams/curves of forces of pressure, which effect on nozzle liner from within and outside with work under the conditions of overexpansion, and graph of change in gas pressure along the length of nozzle.

Fig. 5.7. Diagrams/curves of forces of pressure, which effect on nozzle liner and imaginary extension of nozzle from within and outside with work under the conditions of underexpansion, and graph of change in gas pressure along the length of nozzle.

Page 86.

After the gas velocity in nozzle throat will achieve the local velocity of sound, sharply changes the course of processes in the expanding section of nozzle; gas velocity after the passage of

critical cross section does not descend, but it grows. Nozzle begins to develop thrust.

Further decrease of the pressure of the atmosphere no longer affects the gas flow through the nozzle, but it continues to affect the thrust: it increases/grows with the decrease of the pressure of the atmosphere until the pressure of gas p_c becomes equal to p_h.

In proportion to further decrease of pressure p_h the after-expansion of gas occurs out of the solid nozzle liners. However, the engine thrust continues to grow due to the decrease of product $f_c p_h$ in the formula of thrust and reaches greatest value $P_п$ (when $p_h=0$, i.e. in the vacuum).

Let us examine altitude performances of two engines, which are distinguished only by nozzle design altitude, moreover

$$\bar{f}_{c1} > \bar{f}_{c2} \text{ and } h_{расч1} > h_{расч2}$$

($h_{расч}$ - rated altitude for the engine nozzle).

The thrust of first engine (with the greater nozzle design altitude) is more in the vacuum and it is less in the Earth than in the second engine. Consequently, the thrust of engine, which has nozzle with the larger height, changes with the climb more sharply than the thrust of engine, which has nozzle with the smaller height.

This special feature/peculiarity of altitude performances is explained by reduction/descent examined above in the engine thrust in proportion to the deviation of the mode/conditions of the work of nozzle from the calculated.

Most advantageous from the point of view of obtaining the greatest specific impulse is the nozzle, which ensures condition $p_c=p_h$ at any flight altitude. This nozzle is called nozzle with the ideally variable height.

So that during the stable operation of engine with an increase in flight altitude nozzle would continue to work in the design conditions, it is necessary to increase nozzle design altitude (expansion ratio $\bar{f}_c$), moreover with the work in the vacuum must be provided condition $\bar{f}_c \to \infty$.

Altitude performance of engine with ideal nozzle examined above is the envelope of altitude performances of motor line, which are characterized by only value $\bar{f}_c$ (Fig. 5.8). Altitude performance of this engine passes through the points, which correspond to design conditions of the work of each engine from the series in question (in particular, through points $P_{h\,расч\,1}$ and $P_{h\,расч\,2}$).

Page 87.

By example Laval nozzle with the gradually variable height is the two-position nozzle of telescopic construction/design (Fig. 5.9). With transport and work of engine at low altitudes the mobile section of nozzle is located above exit section of the fixed section of nozzle, which provides compactness of engine, moreover $f_c = f_{c1}$. At the large height/altitude on the special command/crew with the aid of the hydraulic drive the mobile section of nozzle is displaced to end lower position (for exit section of the fixed section of nozzle). In this case the nozzle exit area and, consequently, also value $\bar{f}_c$, respectively increase/grow. Height/altitude h', at which must be given command for the displacement/movement of the mobile section of nozzle for an increase in value $\bar{f}_c$ can be determined from the graph, shown in Fig. 5.8.

Altitude performance of this two-position nozzle is the curve *ВГД* with the fracture at point *Г* (cm Fig. 5.8). Shaded area depicts gain in the thrust in comparison with engines 2 (area A) and 1 (area B), that have nozzles with an invariable nozzle exit area of f_{c2} and f_{c1} respectively.

The example of the nozzle the expansion ratio of which can be smoothly changed, is nozzle with the shaped needle (Fig. 5.10).

Throat area changes in the constant nozzle exit area by displacing the shaped needle within the chamber/camera.

In the overwhelming majority of thermal rocket engines the temperature of gases is sufficiently high. Because of the need for the essential complication of engine the nozzles not only with the smooth, but also with the stepped variation expansion ratio in the contemporary engines in practice do not use.

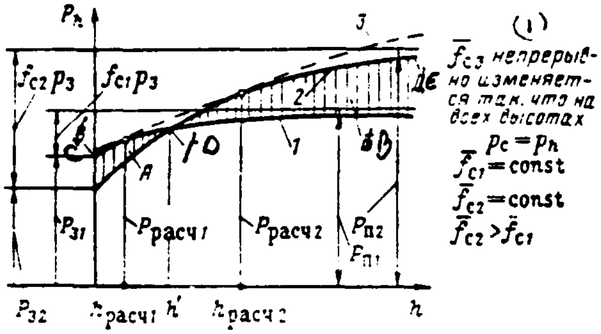


Fig. 5.8. Altitude performances thermal RD with different invariable nozzle design altitude (1, 2) and altitude performance thermal RD with ideally variable height (3).

Key: (1). continuously it changes so that at all heights/altitudes.

Page 88.

§5.3. Procedure of the approximate computation of chamber/camera.

In the series/number of handbooks for the working medium/propellants (combustion products) are given the values β and n_p, which in the approximate computations can be with certain error accepted as constants. Then calculation of chamber/camera is obtained sufficiently to simple ones.

Equation (1.5) taking into account relationship/ratio (4.14) can be written in the following form:

$$P_{n}=K_{p}\beta\dot{m}. \qquad (5.10)$$

During calculation of engine thrust P_{n} is the assigned magnitude. Therefore for calculating the expenditure/consumption of $\dot{m}$ at the known value β it is necessary to determine only coefficient K_{p}.

It is possible to show that value K_{p} with the prescribed/assigned working medium/propellant (prescribed/assigned propellant components) and the prescribed/assigned expansion ratio $\bar{f}_{c}$ depends only on values ε_{c} and $\bar{n}_{p}$.

If we substitute values of $\dot{m}$ and W_{c} from equations (4.14), (4.15) and (4.23) into formula (1.2) and lot of relationship/ratio (4.10), (4.12), (4.16) and (5.10), then after some algebraic conversions we will obtain

$$K_{p}=\sqrt{\frac{2n_{p}^{2}}{n_{p}-1}\left(\frac{2}{n_{p}+1}\right)^{(n_{p}+1)/(n_{p}-1)}\left(1-\frac{1}{\varepsilon_{c}^{(n_{p}-1)/n_{p}}}\right)}+\frac{\bar{f}_{c}}{\varepsilon_{c}}. \qquad (5.11)$$

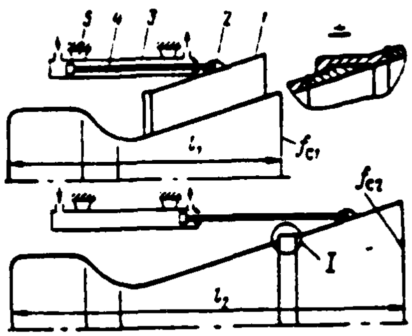


Fig. 5.9.

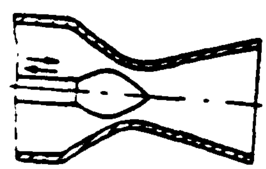


Fig. 5.10.

Fig. 5.9. Two-position nozzle of telescopic construction/design 1 - extensible section of nozzle; 2 - bracket for fastening of stock/rod of hydro-wire/hydro-conductor; 3 - cylinder of hydro-wire/hydro-conductor; 4 - stock/rod of hydro-wire/hydro-conductor; 5 - mounting bracket of hydro-wire/hydro-conductor; l_1 and l_2 - length of chamber/camera before and after advancement of section of nozzle.

Fig. 5.10. Nozzle with shaped needle.

Page 89.

In §4.5. it was noted, that necessary nozzle expansion ratio $\bar{f}_c$ depends only on n_p and ε_c; therefore thrust coefficient in the vacuum

is their function, i.e.,

$$K_p = f(n_p; \varepsilon_c).$$

Values K_P for n_p from 1.08 to 1.25 and for ε_c from 20 to 4500 are given in appendix 4.

According to prescribed/assigned thrust P_n and obtained expenditure/consumption $\dot{m}$ it is possible to determine specific impulse in the vacuum:

$$I_{va\,n} = \frac{P_n}{\dot{m}}.$$

Area f_{kp} is found, using equation (1.5):

$$f_{kp} = \frac{P_n}{K_P p_k}, \qquad (5.12)$$

while area f_c - according to equation (4.12):

$$f_c = \bar{f} f_{kp},$$

moreover value $\bar{f}_c$ is determined on equation (4.13) and appendix 3. After accomplishing of calculations indicated it is possible to find P_3 and $I_{y\pi.3}$ from equations (1.12) and (1.23).

Is examined below the example of the approximate computation of the chamber/camera of oxygen-hydrogen ZhRD.

Example. To determine values m, $I_{y\pi.3}$, P_3 and $I_{va\,n}$ for oxygen-hydrogen ZhRD, if are prescribed/assigned the following parameters: $P_n = 100$ kn, $p_k = 100$ bar and $p_c = 0.5$ bar.

Solution 1. We select approximate values n_p and β for the fuel/propellant of oxygen+hydrogen, using Tables 10.5:

$n_p = 1{,}232$; $\beta = 2431{,}1$.

2. We design coefficient K_p in appendix 4. For $\varepsilon = p_\kappa/p = 100/0.5 = 200$ when $n = 1.23$ we have $K_p = 1.7903$, also, when $n_p = 1.24$ — $K_p = 1.7807$.

Considering change K_p linear in the range of change n_p from 1.23 to 1.24, we obtain for $n_p = 1{,}232$

$$K_p = 1{,}7903 - \frac{(1{,}7903 - 1{,}7807)\cdot 0{,}002}{0{,}01} = 1{,}7884.$$

3. We determine area $f_{\kappa p}$:

$$f_{\kappa p} = \frac{P_n}{K_p p_\kappa} = \frac{100\cdot 10^3}{1{,}7884\cdot 100\cdot 10^5} = 0{,}005592 \text{ м}^2 = 5592 \text{ мм}^2.$$

4. We find expansion ratio $\overline{f}_c$ through appendix 3 for $n_p = 1{,}23$ and $n_p = 1{,}24$ value $\overline{f}_c$ is equal to 18.745 and 18.285 respectively.

Therefore for $n_p = 1{,}232$

$$\overline{f}_c = 18{,}745 - \frac{(18{,}745 - 18{,}285)\cdot 0{,}002}{0{,}01} = 18{,}745 - 0{,}0920 = 18{,}6530.$$

Consequently,

$$f_c = \overline{f}_c f_{\kappa p} = 18{,}6530\cdot 5592 = 104\,308 \text{ мм}^2.$$

Page 90.

5. We determine expenditure/consumption of $\dot{m}$:

$$\dot{m} = \frac{P_{п}}{K_{p}\beta} = \frac{100 \cdot 10^{3}}{1,7884 \cdot 2431,1} = 23,00 \ \overset{(1)}{кг/сек.}$$

Key: (1). kg/s.

6. We design specific impulse in vacuum:

$$I_{уд.п} = \frac{P_{п}}{\dot{m}} = \frac{100 \cdot 10^{3}}{23,00} = 4347,8 \ \overset{(1)}{н \cdot сек/кг.}$$

Key: (1). N•s/kg.

7. We find thrust and specific impulse at the level of sea ($p_{h}=1.013$ bar): $P_{з}=P_{п}-p_{з}f_{с}=100\,000-1.013 \cdot 10^{5} \cdot 104\,360 \cdot 10^{-6}=89\,428$ N:

$$I_{уд.з} = \frac{P_{з}}{\dot{m}} = \frac{89\,428}{23,00} = 3888,2 \ \overset{(1)}{н \cdot сек/кг.}$$

Key: (1). N•s/kg.

§5.4. Effect of the basic calculated parameters on the specific impulse and the sizes/dimensions or chamber/camera.

In §5.1 is examined the altitude effect of the flight of rocket vehicle and flow rate per second to the thrust and the specific impulse of thermal rocket engines. Shows below the effect of different calculated parameters on the specific impulse and the sizes/dimensions of thermal RD.

Complex β affects the engine thrust in the directly proportional dependence. In §4.5 it was noted, that the complex β is thermodynamic propellant property; it only to the low degree depends on values $p_к$ and $\varkappa$.

It is possible to show that when n_p=const relation $f_{кр}/P_п$ does not depend on value β. Actually/really, if we consider equations (4.14) and (5.10), we will obtain

$$\frac{f_{кр}}{P_п}=\frac{m\beta}{p_к m\beta K_p}=\frac{1}{p_к K_p}.$$

Index n_p changes during the utilization of different working medium/propellants (propellant components) in a comparatively low range (see Table 10.5). Therefore taking into account the fact that relation $f_{кр}/P_п$ does not depend on complex β, it is possible to draw the conclusion/derivation that all types thermal RD of one and the same thrust are approximately/exemplarily identical with respect to sizes/dimensions.

The effect of index n_p or k on the sizes/dimensions of nozzle (to degree of widening $\bar{f}_c$) is examined into §4.5. Let us additionally note that with an increase in index n_p the lateral divergent section surface analogous with value $\bar{f}_c$ is decreased, which leads to the

appropriate decrease of the mass of nozzle and simplifies the problem of its cooling.

The effect of index n_p on coefficient K_p can be evaluated, examining appendix 4. The effect indicated is relatively small: with change n_p from 1.03 to 1.25 coefficient K_p is decreased by 6o/o when $\varepsilon_c=20$ and by 16o/o when $\varepsilon_c \doteq 1000$.

Page 91.

Pressure $p_к$ noticeably affects the specific impulse and the sizes/dimensions of chamber/camera thermal RD. With an increase in pressure $p_к$ to 80-100 bars [~80-100 kgf/cm²] at invariable pressure p_c specific pulse substantially is increased, but with further increase in pressure $p_к$ specific impulse increases/grows less noticeably, and in this case increased values $p_к$ are selected to the greater degree for the decrease of sizes of nozzle and chamber/camera as a whole, than for an increase in the specific jet firing.

In spite of the need for an increase in the expansion ratio $\bar{f}_c$ at an increase in pressure $p_к$ and invariable pressure p_c the absolute sizes/dimensions of nozzle (first of all, area f_c and $f_{кр}$) are decreased. The special feature/peculiarity indicated can be explained, using equation (4.14).

With an increase in pressure $p_к$ area $f_{кр}$ is decreased approximately/exemplarily in the inversely proportional dependence. Complex β with increase $p_к$ insignificantly increases/grows. Due to the effect of pressure on the specific impulse for the creation of one and the same thrust with an increase in pressure $p_к$ is required the smaller expenditure/consumption of working medium/propellant (propellant components) $\dot{m}$, which leads, as can be seen from equation (4.14), to the additional reduction in area $f_{кр}$.

With the decrease of pressure p_c increases/grows the specific impulse of chamber/camera but simultaneously they are increased the sizes/dimensions of nozzle (discharge area f_c and the lateral divergent section surface), which is undesirable as a result of an increase in the mass of nozzle and increase of the difficulties of its cooling.

The selection of optimum values $p_к$ and p_c depends on type and special features/peculiarities of thermal RD (see §9.2 and 6.2).

In a number of cases instead of one chamber/camera in the thermal rocket engines they use several chambers/cameras, which have the same gross thrust. The effect of the division of unit chamber/camera into n of chambers/cameras is manifested in the fact that during the division indicated also increases/grows their lateral surface.

Chapter VI.

NOZZLES OF THE CHAMBERS/CAMERAS OF THERMAL ROCKET ENGINES.

The nozzle of chamber/camera thermal RD converts gas enthalpy (combustion products or decomposition/expansion, heating products) into the kinetic jet energy.

During the design of nozzles it is necessary at prescribed/assigned pressure p_c to reduce to a minimum of loss in the nozzle and to ensure for prescribed/assigned areas $f_{\text{кр}}$ and f_c least possible length and the lateral surface of nozzle, which gives the series/number of advantages, namely:

a) is decreased the mass of nozzle;

b) is facilitated cooling nozzle and chamber/camera as a whole;

c) is decreased the hydraulic resistance of the coolant passage of nozzle (if it is cooled by the working medium/propellant, which takes place between the double nozzle liners).

Page 92.

During calculation of each concrete nozzle it is important to find its optimal profile/airfoil, and also optimal geometric relationships/ratios.

In form are distinguished circular nozzles and nozzles with the inner body (Fig. 6.1). On the geometric form of divergent section the circular nozzles are subdivided into the conical ones and shaped.

Conical nozzles (Fig. 6.2) satisfy with angle $2\theta_c$ by the equal to approximately/exemplarily 25-30°. For prescribed/assigned nozzle expansion ratio with an increase in angle $2\theta_c$ is decreased the length of nozzle and the surface of its walls, but increase/grow losses to the nonparallelism of the streams of the flow of gas of the longitudinal axis of nozzle. Conical nozzles are most simple in the manufacture, but they in the magnitude of losses and to mass characteristics are inferior to shaped nozzles.

Shaped nozzles (Fig. 6.3) are the most widely used type of the nozzles of chambers/cameras thermal RD. They have variable along the length of divergent section expansion angle 2θ: greatest in the area of critical cross section and smallest in exit section. Such a geometry provides the essential advantages of the shaped nozzles

before conical: with one and the same thrust coefficient K_P length and the mass of shaped nozzles, and also the surface of their walls is approximately/exemplarily to 30-50% less than in conical ones.

Certain shortcoming in the shaped nozzles is the relative complexity of their manufacture.

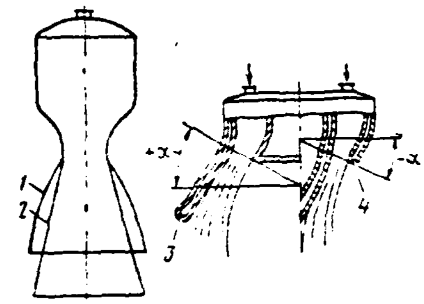


Fig. 6.1.

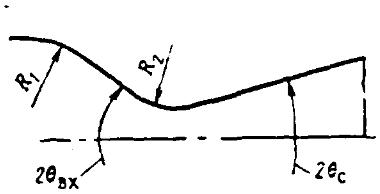


Fig. 6.2.

Fig. 6.1. Types of nozzles of chambers/cameras RD: 1 - shaped; 2 - conical; 3 - nozzle of internal expansion (with skirt): 4 - nozzle of external expansion with inner body in the form of cone; +α and -α - positive and negative angles of surface slope of critical cross section.

Fig. 6.2. Duct/contour of conical nozzle of chamber/camera.

Page 93.

The characteristic feature of nozzles with the inner body is the fact that their critical cross section is ring.

The nozzle throat plane with the inner body in the particular case can be arranged/located perpendicular to the axis/axle of the

chamber/camera (it is analogous with circular nozzle). But greatest interest are of the so-called nozzles with the forced deviation of flow. The plane of their critical ones of cross section is arranged/located at certain angle (positive or negative) to the axis/axle of chamber/camera (cm Fig. 6.1), which causes the deviation of flow to the axis/axle of nozzle (to the inner body) or from it (to the external nozzle liner, called skirt). By changing the angle indicated it is possible to obtain different of the distribution of thrust between the inner body and the skirt.

If the flow of gas diverges to the side of inner body and entire/all thrust is created by inner body, then skirt can be removed, in this case the flow of gas, which escapes behind the nozzle, from the outer side will come into contact with the environment. Such nozzles call nozzles with the external expansion. Inner body is complete or truncated cone.

If the flow of gas diverges to the side of skirt, and entire/all thrust is created by skirt, that inner body it is expedient to satisfy by short. The flow of gas is forced against skirt and occupies only peripheral volume in the walls of skirt. In this case with the environment touches the internal surface of the flow of gas. Such nozzles call nozzles with the internal expansion.

Nozzles with the inner body possess the essential advantages in front of the circular nozzles. Thrust, arriving per unit of the surface of inner body or skirt (especially near the critical cross section where their surface is almost perpendicular to the axis/axle of chamber/camera), it is considerably more than the thrust of circular nozzles. Therefore the nozzles with the inner body to the crust of circular ones whose walls are arranged/located at sufficiently small angle to the axis/axle and must have large surface for the creation of one and the same thrust.

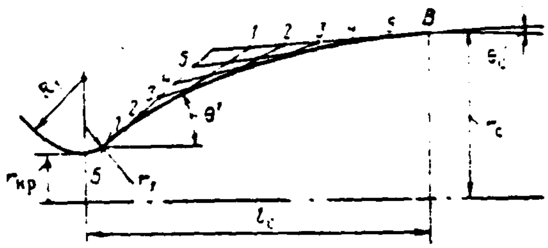


Fig. 6.3. Construction of the duct/contour of the shaped nozzle of chamber/camera.

Page 94.

The important advantage of nozzles with the inner body is the automatic regulation of expansion ratio. During the off-design conditions of the work of nozzle the flow of gas changes its volume. With the work under the conditions of underexpansion the flow of gas is forced against skirt in the nozzle with the internal expansion or against the inner body in the nozzle with the external expansion. The possibility of the overexpansion of flow gas in such nozzles virtually is eliminated: gas is expanded only up to a pressure of the environment, after which it blows away from the inner body or the skirt. After all with form, nozzles with the inner body provide the optimal expansion ratio of gas and high specific impulse with the work of engine at any height/altitude.

Nozzles with the inner body are located in the stage of investigations, and for their introduction in the real constructions/designs is in prospect of overcoming the series/number of difficulties - first of all to solve the problem of cooling. Chambers/cameras with the nozzles indicated possess in comparison with usual type chambers/cameras greater surface which must be cooled. Nozzles with the inner body have considerably greater perimeter of critical cross section; as it will be shown in Chapter VI, in the region this cross section heat fluxes from the gas to the wall have the greatest values. Furthermore, in the chambers/cameras indicated hinders the delivery of coolant to the surfaces of the walls which must be cooled.

§6.1. Losses in the nozzles. The selection of form and expansion angles puffed.

The forms of losses in the nozzles were given into §4.6. Let us examine the losses indicated in more detail [17].

The losses, caused by nonparallel nature of the streams of the working axle center of the nozzle (see Fig. 1.6). In parallel to the axis/axle of nozzle flow out only the streams of the working

medium/propellant, arranged/located about the axis/axle of nozzle. Remaining streams flow out behind the nozzle at certain angle to its axis/axle, the angle increasing/growing in proportion to removal/distance from the axis/axle of nozzle.

The dependence of the coefficient, which characterizes the losses indicated (let us designate its φ_1) from angle θ_c is expressed by the equation

$$\varphi_1 = \frac{1+\cos\theta_c}{2}. \qquad (6.1)$$

Are given below values φ_1 for some angles $2\theta_c$:

$2\theta_c$	12	16	20	24	28	32	36	40
φ_1	0,9972	0,9951	0,9924	0,9890	0,9851	0,9806	0,9755	0,9698

Formula (6.1) is valid for the conical nozzle. However, it is possible to use, also, for the shaped nozzles, since coefficient φ_1 for them depends in essence on angle θ in nozzle exit section, i.e., from angle θ_c; therefore for the shaped nozzles

$$\varphi_1 \approx \frac{1+\cos\theta_c}{2}.$$

Page 95.

Wall-friction loss of nozzle. With the flow of working medium/propellant about the nozzle liners appears the force, caused

by the presence of the friction of working medium/propellant against the walls. It decreases to a certain extent the thrust force of chamber/camera. For prescribed/assigned areas $f_{кр}$ and f_a the losses to the friction descend with the decrease of the length (and, consequently, of surface) of nozzle, and also with an improvement in the purity/finish of the treatment of the internal surface of their walls. Wall-friction loss are estimated by coefficient ϕ_2; for the nozzles ZhRD ϕ_2=0.980-0.995.

Entry loss into the nozzle. Above it was assumed that gas reaches critical speed ($W=a_{кр}$) strictly in the throat plane. In reality as a result of the compression of gas jets during their motion along the tapering portion of the nozzle the pressure in the central streams is more than about its wall. Therefore gas in nozzle liners earlier is accelerated/dispersed to the critical speed and crosses it over the curved surface, which by its convexity is turned to the side of expanding section of nozzle. With excessive camber in the critical cross section, and also in the region of coupling the ducts/contours with the different curvature can appear the large nonuniformity of the field of the velocities in the critical cross section, shock waves and flow breakaway. The noted special features/peculiarities affect the character of flow in the expanding section of nozzle, causing the appropriate losses. In order to exclude the possibility of the emergence of shock waves, nozzle

contour is satisfied in the form sufficient expanded and smooth curve. Entry loss into the nozzle will be low, if we observe the following relationships/ratios during the construction of nozzle contour (see Fig. 6.2):

1) the radius of coupling the inswept and expanding section of nozzle $R_2=(0,65\div1,5)d_{кр}$;

2) the radius of coupling cylindrical chamber wall with the tapering portion of the nozzle

$$R_1=(0,4\div0,5)d_{кр};$$

3) the angle of the tapering portion of nozzle $2\theta_{вх}=60\div90°$; however frequently the conical section is absent, i.e., radii R_1 and R_2 are coupled with each other.

For the construction of the duct/contour of profiled nozzle are recommended the following relationships/ratios (see Fig. 6.3): $R_1=0,75d_{кр}$ and $r_1=0,225\,d_{кр}$.

Page 96.

The construction of the divergent section of shaped nozzle is examined at the end of the present paragraph.

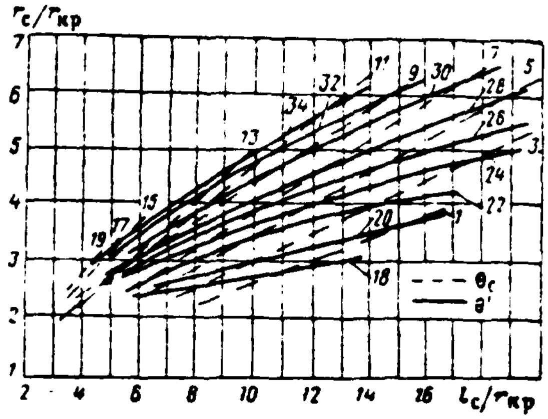
Losses to the inequilibrium of the process of expansion. Depending on the degree of inequilibrium (partially or completely unbalanced expansion) the specific impulse of chamber/camera descends by 5-10 % in comparison with its value for the equilibrium expansion.

Losses, caused by the branch/removal of heat from the working medium/propellant to the walls. These losses depend on the type of cooling chamber/camera. If chamber/camera has the coolant passage, and the taking place on it coolant is supplied then inside the chamber/camera, then the losses of specific impulse as a result of the branch/removal of the heat of working medium/propellant into the walls will not be. If chamber/camera does not have the coolant passage, then the heat fluxes, which come from working medium/propellant the walls, give rise to the appropriate losses of specific impulse; in this case of chamber/camera it forever radiates heat into the surrounding space.

Losses, caused by the formation of the condensed phase to the process of moving the working medium/propellant along the nozzle. In a number of cases, for example with the addition of metals into the fuel/propellant, in the products, which take place on the nozzle, are formed the particles of the condensed phase. A temperature drop and an increase in the particle speed remains from a change in the

corresponding parameters of gas, which conditions the losses of specific impulse; in certain cases these losses can compose 3-10o/o and more.

Expansion angle in exit section of shaped nozzles is selected in limits $2\theta_c$ =20-28°.



Fiq. 6.4. The dependence of anjles θ_c and ϑ' on the values of ratios $l_c/r_{кр}$ and $r_c/r_{кр}$ (numerals on the lines indicate angles in the degrees).

Page 97.

During calculation of angle θ_c of the nozzles thermal RD with the large height, intended for the work in the vacuum, it is possible to use the equation

$$\sin 2\theta_c = \frac{p_c - p_h}{\rho_c W_c^2/2}\sqrt{M_c^2 - 1}, \qquad (6.2)$$

where $M_c = W_c/a_c$.

Radius $r_c = d_c/2$ obtain from the thermal design chambers/cameras.

The missing for the construction of the duct/contour (see Fig.

6.3) values (angle θ' and the length of nozzle l_c) are determined with the aid of the grid (Fig. 6.4).

Nozzle contour in the section from point B to point C is the parabola, construction by which is shown in Fig. 6.3.

§6.2. Selection of optimal pressure

For the first-stage engines of rockets it is expedient to select this gas pressure in nozzle exit section, with which the mode of its operation to the smallest degree deviates from the design conditions on the average along the trajectory of the powered phase (see §5.2).

For the engines, intended for the work in outer space, pressure p_c can be selected as to low ones as desired, and in spite of this rated nozzle conditions will not be secured. It is necessary to indicate that the selection of very low pressure p_c causes under condition p_K=const. on one hand, an increase of the expansion ratio of gas in nozzle ε_c, and, consequently, specific impulse, but on the other hand, increase of the necessary sizes/dimensions of nozzle (value f_c) and its mass. The criterion of the selection of pressure p_c is the characteristic velocity of rocket vehicle or its step/stage.

With an increase in area f_c to value $f_c=f_{c.\text{опт}}$ (Fig. 6.5) and the corresponding decrease of pressure p_c growth in the specific impulse covers the effect of an increase in the mass of nozzle by the characteristic velocity, so that the latter increases/grows. With further increase in area f_c and corresponding decrease of pressure p_c the mass of nozzle increases/grows to the greater degree than is increased specific impulse, as a result of which the characteristic velocity is decreased.

The effect of a change in the specific impulse and mass of engine on the characteristic velocity of vehicle is convenient to estimate so called mass equivalent of specific impulse.

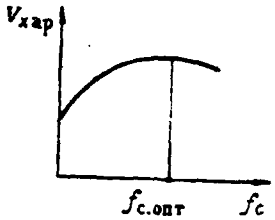


Fig. 6.5. Dependence of the characteristic velocity of rocket vehicle with thermal RD on area f_c

Page 98.

Let us write the equation of Tsiolkovskiy for the work of engine in the vacuum when $V_{нач}=0$, using relationships/ratios (2.1) and (1.22) and after designating relation $m_{нач}/m_{кон}$ through $\mu_{кон}$:

$$V_{кон}=I_{уд.п}\ln\mu_{кон}.$$

Let us differentiate the obtained equation, counting $V_{кон}$ =const:

$$0=dI_{уд.п}\ln\mu_{кон}+I_{уд.п}\frac{d\mu_{кон}}{\mu_{кон}}$$

$$\text{or}\ \frac{d\mu_{кон}}{\mu_{кон}}=-\ln\mu_{кон}\frac{dI_{уд.п}}{I_{уд.п}}.$$

Let us write the latter/last equation through the increases

$$\frac{\Delta\mu_{кон}}{\mu_{кон}}=-\ln\mu_{кон}\frac{\Delta I_{уд.п}}{I_{уд.п}}. \qquad (6.3)$$

Equation (6.3) is the equation of the equivalent of specific impulse. With its help it is possible to determine, it is expedient to increase specific impulse via the selection of lower pressure p_c, or this increase becomes unfavorable due to increase of mass of nozzle. Tentatively it is possible to consider that to an increase in the specific impulse on one percent is equivalent the decrease of the finite mass of vehicle to 10-15o/o; the value indicated, as can be seen from equation (6.3), it depends on value $\mu_{кон}$ of rocket vehicle.

With the decrease of the mass of the finite segments of nozzle (for example, via failure of the coolant passage with the double walls and the transition/transfer for the cooling by radiation/emission into the surrounding space) it is possible to additionally decrease pressure p_c and to raise specific impulse $I_{уд}$.

In the presence of the coolant passage it is necessary to consider the following shortcomings, connected with the decrease of pressure p_c, with an increase in area f_c and expansion ratio $\bar{f}_c$ (besides the shortcomings, caused by an increase in sizes/dimensions and mass of nozzle): increases/grows hydraulic resistance of the coolant passage of nozzle, and becomes complicated the problem of cooling chamber/camera.

For the nozzles of the engine chambers of the first pressure stage p_c usually selects equal to approximately/exemplarily 0.5-0.7 bars [≈0.5-0.7 kgf/cm²], while for the nozzles of the engine chambers, intended for the work under conditions of outer space, to 0.01-01 bars [≈0.01-0.1 kgf/cm²].

One should emphasize that the engines, which have nozzles with the large height (i.e. with low pressure p_c), cannot be started at the

level of sea, since in this case they will work in the mode/conditions of significant overexpansion; pressure on the final part of the nozzle will be outside more than from within, that it is possible to cause the warping of nozzle or the flow breakaway of gas from its walls. In the latter case decreases the actual value of nozzle expansion ratio $\bar{f}_c$.

Page 99. Chapter VII.

Thermal design of thermal rocket engines.

§ 7.1. Procedure of the thermal design of thermal rocket engines.

During the thermal design of the chamber/camera of thermal rocket engines is determined:

a) the specific impulse in vacuum $I_{уд.п}$;

b) necessary flow rate per second of working medium/propellant (fuel/propellant $\dot{m}$;

c) the necessary sizes/dimensions of the nozzle of chamber/camera - the value of areas $f_{кр}$ and f_{c}.

For accomplishing calculation they must be known:

a) initial is working body (for the chemical rocket engines - propellant components and coefficient x);

b) thrust in vacuum $P_{п}$;

c) the pressure of working medium/propellant at the nozzle entry $p_{\text{к}}$;

d) the pressure of working medium/propellant in nozzle exit section $p_{\text{с}}$.

Actually optimum values $p_{\text{к}}$, $p_{\text{с}}$ and $\varkappa$ are previously unknown: for their selection are necessary special calculations (see § 6.2, 9.2 and 10.6); however, in the thermal design they are considered the given ones.

Working body in the chamber/camera of thermal rocket engines is the mixture of gases in which, as it was noted in chapter IV, can be found substances in the liquid and solid state, and also ions and electrons. Working body in the chamber/camera (for example, at the nozzle entry and at the output from it) in composition and parameters in many respects differs from working medium/propellant by the entry into the chamber/camera (from the initial working medium/propellant).

The procedure of calculation of chamber/camera following: are determined the theoretical values of specific impulse in vacuum $I_{\text{уд.п}\,t}$, the flow rate per second of working medium/propellant $\dot{m}_t$, nozzle

throat area $f_{кр\,t}$ and nozzle exit area f_{ct}. In this calculation consider only the phenomena, connected with the dissociation and the recombination and the losses, connected with the carry-off of heat and chemical energy together with the stream of working medium/propellant, which escapes behind the nozzle.

Then are determined the actual values of the parameters ($I_{уд.п.д}$; $\dot{m}_д$, $f_{кр.д}$ and $f_{с.д}$) indicated taking into account all remaining losses.

Page 100.

The thermal design of chamber/camera thermal RS is conducted in the following sequence.

1. Is found through tables total enthalpy of working medium/propellant $i_{п.вх}$ for conditions for entry into chamber/camera.

In general in the engine chamber to the working medium/propellant it is possible to supply or to abstract/remove from it a quantity of heat Q. Let us designate the total enthalpy of working medium/propellant at the nozzle entry $i_{п.к}$. then in accordance with the law of conservation of the energy

$$i_{п.вх} \pm Q = i_{п.к}. \qquad (7.1)$$

2. Are determined temperature and composition of working medium/propellant at nozzle entry, on the basis of condition, expressed by equation (7.1).

3. Is designed by obtained composition of working medium/propellant its entropy at nozzle entry $s_к$ (see § 7.8).

4. Find temperature and composition of working medium/propellant in nozzle exit section, considering expansion maximally equilibrium (see § 4.4) and isentropic (s=const), i.e., entropy of working medium/propellant at nozzle entry $s_к$ is equal to entropy in its exit section:

$$s_к=s_c. \qquad (7.2)$$

5. Is determined by obtained composition total enthalpy of working medium/propellant in nozzle exit section $I_{п.c}$.

6. Is designed ideal exhaust velocity of working medium/propellant at nozzle outlet, according to equation (4.21), where total enthalpy must be expressed in kJ/kg, and constant number (2000) is carried out from under root;

$$W_c=44{,}72\sqrt{I_{п.к}-I_{п.с}}. \qquad (7.3)$$

7. is found through equation (4.4) the density of working medium/propellant in nozzle exit section

$$\varrho_c=\frac{p_c}{R_cT_c}. \qquad (7.4)$$

Gas constant R_c is designed from equation (4.5), moreover apparent molecular weight μ_g is found in composition of the working medium/propellant (see § 7.9).

8. Is determined according to equation (1.22) theoretical specific impulse in vacuum:

$$I_{\text{уд п } t}=W_{ct}+\frac{f_{ct}}{\dot{m}_t}p_c. \qquad (7.5)$$

The unknown value of relation $f_{ct}/\dot{m}_t$ is determined from equation (4.9):

$$\frac{f_{ct}}{\dot{m}_t}=\frac{1}{\varrho_c W_{ct}}.$$

Page 101.

9. Is designed from equation (1.21) theoretical flow rate of working medium/propellant $\dot{m}_t$, necessary for creation of prescribed/assigned thrust $P_п$:

$$\dot{m}_t=\frac{P_п}{I_{\text{уд.п}t}}. \qquad (7.6)$$

10. Is found through equation (4.9) theoretical nozzle exit area

$$f_{ct}=\frac{\dot{m}_t}{\varrho_c W_{ct}}. \qquad (7.7)$$

11. Is determined according to equation (4.7) or (4.8) index of equilibrium process of expansion n_p.

12. Is designed from equation (4.15) theoretical value of complex β:

$$\beta_t=\frac{\sqrt{R_к T_{кt}}}{A_{n_p}}. \qquad (7.8)$$

Value A_{π_e} is determined according to equation (4.16).

13. Is found through equation (4.14) theoretical value of throat area

$$f_{\text{кр}\,t}=\frac{\dot{m}_t\beta_t}{p_\text{к}}. \qquad (7.9)$$

14. Determine (if necessary) theoretical values of thrust and specific impulse at the level seas P_{3t} and $I_{\text{у.з.}t}$ using equations (1.12) and (1.23).

On this calculation of the chamber/camera of thermal rocket engines without taking into account many losses is finished.

As is evident, in the thermal design determine the thermodynamic parameters (i, s, R) of working medium/propellant in the characteristic cross sections the chambers/cameras. For this it is necessary to design the equilibrium chemical composition of working medium/propellant; into most cases (KhRD, YaRD, ERD) it is sufficiently complex (ten and more than substances). Therefore the determination of composition occupies the large part of the thermal design.

§ 7.2. The total enthalpy of working medium/propellant.

In § 4.1 and 4.5 it was shown that during the motion of working

medium/propellant along the chamber/camera of thermal rocket engines changes not only the enthalpy of the working medium/propellant i, but also its chemical energy. The generalized parameter, which encompasses and enthalpy, and chemical energy of working medium/propellant, is total enthalpy i_n. The concept of the total enthalpy is closely related with the concept of heat of formation of substance, which is called the quantity of heat, isolated or absorbed with the formation of the unit of the mass of this substance.

Page 102.

Heat of formation of substance depends on temperature T which is selected for the reference point. This temperature have parent substances and to it are cooled or are heated the end products of reaction in the branch/removal from them or the delivery to them of the corresponding quantity of heat, which is heat of formation of substance at a temperature T and is designated ΔH_{fT}.

The chemical energy of substance composes basic part of the heat of its formation whose value is the measure for a change of the chemical energy in the reaction of formation of this substance. Strictly speaking, a change in the chemical energy relates to all substances, which participate in the reaction, but for simplification in its calculations conditionally they carry to one of the

substances.

For convenience in calculations of the total enthalpy are introduced the concepts of the standard state of elements/cells and standard heat of formation.

The standard state of elements/cells is called their stable and most widely used in the nature state at a selected temperature T_0, taken as the reference point of enthalpy, and at a pressure of p_0=1.013 bar [1 phys. at.], moreover in the state indicated the total enthalpy of elements/cells takes as equal to zero. Let us accept, as in handbook [15], for the reference point temperature T=293°.15 K. Standard state for hydrogen, oxygen, fluorine and nitrogen with T_0=293°.15 K and p_0=1.013 bars [1 phys. atm.] is phased state in the form of diatomic gas.

Standard heat of formation of substance is called the heat of its formation from the elements/cells, which are found in the standard state; this heat designate $\Delta H^{\circ}_{fT_0}$.

Subsequently it is accepted:

1. A change in the chemical energy with the course of reaction relates to the substance, formed from the standard elements/cells as

a result of reaction.

2. Chemical energy and heat of formation of substance are considered positive, if as result of reaction heat is absorbed by substance, and negative - if heat by them is allotted.

The total enthalpy of substance is equal to the sum of its chemical energy, evaluated by heat of formation, and enthalpy. Therefore taking into account the noted above special features/peculiarities it is possible to write the following formula of the total enthalpy at an arbitrary temperature T, expressed through the standard heat of formation:

$$I_{n\,T}=\Delta H^0_{fT_0}+\int\limits_{T_0}^{T} c^0_p\,dT$$

or taking into account equation (4.20)

$$I_{n\,T}=\Delta H^0_{fT_0}+(I^0_{n\,T}-I^0_{n\,T_0}). \qquad (7.10)$$

Page 103.

Obtained experimentally values $\Delta H^0_{fT_0}$ are brought to tables [15], whence it is possible to take them for calculations.

A difference in the total enthalpies of substance at temperatures T and T_0 is determined in the most general case according to the formula

$$I^0_{n\,T}-I^0_{n\,T_0}=\int\limits_{T_0}^{T}\sum c^0_i\,dT+\sum\Delta H^0_i. \qquad (7.11)$$

where c^0 - heat capacity of substance in the gaseous (c_p^0), liquid $(c_ж^0)$ and solid $(c_{тв}^0)$ state;

ΔH_f^0 - heat of phase transformations (melting and vaporization) and polymorphic transformations of substance.

The heat of phase and polymorphic transformations ΔH_i and heat capacity $c_ж^0$ and $c_{тв}^0$ is determined experimentally; heat capacity is phase c_p^0 they design [15].

Assuming that heat capacities $c_ж^0$ and $c_{тв}^0$ do not depend on temperature, then a difference in the total enthalpies for the substance, which at a temperature T_0 is located in the solid, but at a temperature T - in the liquid state, it is equal to

$$i_{nT}^0 - i_{nT_0}^0 = c_{тв}^0(T_{пл} - T_0) + c_ж^0(T - T_{пл}) + \sum \Delta H_i^0. \quad (7.12)$$

The values of the total enthalpy of basic oxidizers and fuels are given in tables 10.3 and 10.4.

For the cryogenic elements (for example, H_2, O_2 and F_2) the total enthalpy is different from zero, while under the standard conditions it is equal to zero. This is easy to explain: hydrogen, oxygen and fluorine under the standard conditions are gases; for the liquefaction of these substances it is necessary to lead of them heat in order to decrease the temperature of gas to $T = T_{кип}$ and to ensure

the phase transformation (condensation).

The value of the total enthalpy 1 kg. of working medium/propellant nonchemical thermal RD, monopropellant and component two- and composite propellants, which are the mixture of several individual substances which are dissolved in each other, is designed from the formula

$$i_{\text{п}}=\sum_{i=1}^{i=n} g_i i_{\text{п}\,i} \pm \sum g_{i\,\text{рас}} Q_{i\,\text{рас}}, \qquad (7.13)$$

where g_i and $i_{\text{п}i}$ - mass portion and the total enthalpy of the i individual substance; $g_{i\,\text{рас}}$ - mass portion of the i solvend: $Q_{i\,\text{рас}}$ - heat of solution 1 kg. of the i solvend in the complex solvent.

Positive sign in equation (7.13) indicates ingress of heat, while sign "minus" - its liberation.

Page 104.

If individual substances in the mixture are not dissolved in each other ($Q_{i\,\text{рас}}=0$), then formula for calculating the total enthalpy 1 kg. of this mixture takes the following form:

$$i_{\text{п}}=\sum_{i=1}^{i=n} g_i i_{\text{п}\,i}. \qquad (7.14)$$

The total enthalpy for two-component fuel chemical RD is

determined from formula, analogous formula (7.30) (see § 7.5):

$$i_{п.т}=\frac{i_{п.г}+\varkappa i_{п.ок}}{1+\varkappa}. \qquad (7.15)$$

The total enthalpy of the mixture of gases (combustion products, decomposition or heating) depends on their composition and temperature and is designed from the formula

$$i_{п}=\sum_{i=1}^{i=n} M_i i_{п\,i}, \qquad (7.16)$$

where M_i - number of kilomoles of the i gas in 1 kg. of the mixture of gases at its temperature and pressure.

Let us designate a number of kilomoles in 1 kg. of the mixture of the gases through M_Σ and let us write the following relationships/ratios, known from the thermodynamics:

$$M_\Sigma=\sum_{i=1}^{i=n} M_i; \qquad (7.17)$$

$$M_i=p_i\frac{M_\Sigma}{p_\Sigma}; \qquad (7.18)$$

$$\frac{M_\Sigma}{p_\Sigma}=\frac{1}{\sum_{i=1}^{i=n}\mu_i p_i}. \qquad (7.19)$$

With their account equation (7.16) can be written in the following forms:

$$i_{п}=\frac{M_\Sigma}{p_\Sigma}\sum_{i=1}^{i=n} i_{п\,i}\,p_i; \qquad (7.20)$$

$$i_{п}=\frac{\sum_{i=1}^{i=n} i_{п\,i}\,p_i}{\sum_{i=1}^{i=n}\mu_i p_i}, \qquad (7.21)$$

where $i_{\text{п}i}$ - the total enthalpy or the i gas which is taken from tables [15].

Page 105.

§ 7.3. Mass composition of working medium/propellant.

The composition of working medium/propellant is expressed by the mass portions of equivalent components:

$$g_{\text{э}_j}=\frac{n_{\text{э}_j}A_{\text{э}_j}}{\mu}, \qquad (7.22)$$

where $g_{\text{э}_j}$ - mass portion of the j element/cell; $n_{\text{э}_j}$ - number of atoms of the j element/cell in the molecule of working medium/propellant; $A_{\text{э}_j}$ - atomic mass of the j element/cell; μ - molecular weight of working medium/propellant.

The mass portion of element/cell they designate by its index; for example, $H_{\text{ок}}$ - the mass portion of hydrogen in the oxidizer, and $H_{\text{г}}$ - in the fuel.

Molecular weight of working medium/propellant is determined from the formula

$$\mu=\sum_{j=1}^{j=n} n_{\text{э}_j}A_{\text{э}_j}. \qquad (7.23)$$

During the check of calculations is used the relationship/ratio

$$\sum_{j=1}^{j=n} g_{\text{э}_j}=1.$$

Example 1. To calculate the mass composition of nitrogen tetroxide N_2O_4 of 100o/c concentration.

Solution. We find through formula (7.23) the molecular weight of nitrogen tetroxide

$$\mu=2\cdot14{,}007+4\cdot15{,}999=92{,}010.$$

We determine from formula (7.22) the mass portions N and O:

$$N_{ox}=\frac{2\cdot14{,}007}{92{,}010}=0{,}304;\quad O_{ox}=\frac{4\cdot15{,}999}{92{,}010}=0{,}696.$$

We check the mass composition:

$$N_{ox}+O_{ox}=0{,}304+0{,}696=1{,}0.$$

Calculation of mass composition somewhat becomes complicated, if working body is the mixture of several individual chemical substances. In this case calculation they conduct according to the formula

$$g_{эj}=\sum_{i=1}^{i=n}g_i g_{ji}, \qquad (7.24)$$

where g_i - mass portion of the i substance in the working medium/propellant; g_{ji} - mass portion of the j element/cell in the i-th of substance; it are determined from formula (7.22).

Page 106.

Example 6. To calculate the mass composition of aerozine-50, which is the mixture 50o/o hydrazine N_2H_4 and 50o/o of unsymmetrical dimethylhydrazine (UDMH).

Solution. We determine the mass composition of N_2H_4 by formulas (7.22) and (7.23):

$$\mu_{N_2H_4} = 2\cdot 14{,}007 + 4\cdot 1{,}008 = 32{,}046;$$

$$H_{г1} = \frac{4\cdot 1{,}008}{32{,}046} = 0{,}126; \quad N_{г1} = \frac{2\cdot 14{,}007}{32{,}046} = 0{,}874.$$

We find the mass composition UDMH, which has a formula $H_2N{-}N(CH_3)_2$:

$$\mu_{НДМГ} = 2\cdot 14{,}007 + 8\cdot 1{,}008 + 2\cdot 12{,}011 = 60{,}1;$$

$$C_{г2} = \frac{2\cdot 12{,}011}{60{,}1} = 0{,}400; \quad H_{г2} = \frac{8\cdot 1{,}008}{60{,}1} = 0{,}134;$$

$$N_{г2} = \frac{2\cdot 14{,}007}{60{,}1} = 0{,}466.$$

We calculate the mass composition of fuel by formula (7.24):

$$N_г = g_{N_2H_4}\cdot N_{г1} + g_{НДМГ} N_{г2} = 0{,}5\cdot 0{,}874 + 0{,}5\cdot 0{,}466 = 0{,}67;$$

$$H_г = g_{N_2H_4}\cdot H_{г1} + g_{НДМГ}\cdot H_{г2} = 0{,}5\cdot 0{,}126 + 0{,}5\cdot 0{,}134 = 0{,}13;$$

$$C_г = g_{НДМГ} C_{г2} = 0{,}5\cdot 0{,}4 = 0{,}20.$$

We check the mass composition of fuel:

$$N_г + H_г + C_г = 0{,}67 + 0{,}13 + 0{,}20 = 1.$$

§ 7.4. Coefficients $\varkappa$ and $\alpha_{ок}$

The coefficient of the real relationship/ratio of components of propellant $\varkappa$ chemical RD is called the relation of the mass oxidizer

consumption and fuel per second:

$$\varkappa=\frac{\dot m_{ox}}{m_r}. \qquad (7.25)$$

Oxidizer fuel can be supplied into the chamber/camera ZhRD in different relationships/ratios, in this case in the fuel/propellant and, consequently, also in the combustion products can be contained the excess of oxidizer or fuel. Fuels/propellants RDTT and RDGT also can have an excess of oxidizer or fuel. Combustion products with the significant excess of oxidizer call oxidative gas, and with the significant excess of fuel - by reducing gas.

Furthermore, is possible the so-called stoichiometric relationship/ratio of components of propellant $\varkappa_{ctex}$, with which the quantity of oxidizer, which falls on 1 kg. of fuel, in the precision/accuracy is equal to the quantity necessary for its complete oxidation.

Let us derive formula for calculating the coefficient $\varkappa_{ctex}$ for the fuel/propellant, which consists of four chemical elements: C, H, O and N.

Page 107.

In general the composition of fuel is expressed by mass portions C_r, H_r, O_r and N_r, and the composition of oxidizer - by mass portions

$C_{ок}$, $H_{ок}$, $O_{ок}$ and $N_{ок}$.

Let us determine the small quantity of oxidizer (oxygen), necessary for the complete oxidation of carbon and hydrogen, using for this purpose of the equation of the burning

$$C + O_2 = CO_2, \qquad (7.26)$$

$$H_2 + 1/2\ O_2 = H_2O. \qquad (7.27)$$

Express is mass quantities of elements/cells, entering equations (7.26) and (7.27), in the kilcmcles or the kilo-atoms. One kilomole of oxygen and hydrogen is 31.999 and 2.016 kg. respectively, while one kilomcle of carbon dioxide and water vapor - 44.01 and 18.015 kg. respectively. One kilo-atom of carbon is equal to 12.011 kg. Entering/writing a mass quantity of elements/cells in the equation of burning (7.26), we obtain

$$12{,}011\ кг\ C + 31{,}999\ кг\ O_2 = 44{,}01\ кг\ CO_2;$$
$$1\ кг\ C + 31{,}999/12{,}011\ кг\ O_2 = 44{,}01/12{,}011\ кг\ CO_2;$$
$$1\ кг\ C + 2{,}664\ кг\ O_2 = 3{,}664\ кг\ CO_2.$$

In 1 kg. of fuel is contained C_r kg. of carbon. Therefore

$$C_r\ кг\ C + 2{,}664\ C_r\ кг\ O_2 = 3{,}664\ C_r\ кг\ CO_2.$$

It is analogous for the equation of burning (7.27)

$$2{,}016\ кг\ H_2 + {}^1/_2 \cdot 31{,}999\ кг\ O_2 = 18{,}015\ кг\ H_2O;$$
$$1\ кг\ H_2 + 7{,}937\ кг\ O_2 = 8{,}937\ H_2O;$$
$$H_r\ кг\ H_2 + 7{,}937\ H_r\ кг\ O_2 = 8{,}937\ H_r\ кг\ H_2O.$$

Consequently, for complete oxidation C_r of the kilograms of carbon and H_r the kilograms of hydrogen is necessary with respect 2.664 C_r and 7.937 H_r the kilograms of oxygen.

If in 1 kg. of fuel is contained O_r kg. of oxygen, then for the complete oxidation 1 kg. of fuel it is necessary to bring not (2.664 C_r +7.937 H_r) kg. of oxygen, but are less to value O_r, (2.664 C_r +7.937 H_r—O_r) kg. of oxygen. If oxidizer is pure/clean oxygen, then calculation is completed. However, oxilizer can contain hydrogen and carbon, and also nitrogen.

A quantity of oxygen in the oxidizer is determined by mass portion O_{ok}. The part of oxygen from the composition of oxidizer will be expended/consumed on the oxidation of carbon and hydrogen, that are in its composition, in this case for oxidation C_{ok} kg. of carbon is required 2.664 C_{ok} kg. of oxygen, and for oxidation H_{ok} kg. of hydrogen - 7.937 H_{ok} kg. of oxygen.

Consequently, in 1 kg. of oxidizer is contained (O_{ok}—2,664 C_{ok}—7,937 H_{ok}) kg. of free oxygen.

Page 108.

After the calculations indicated it is possible to calculate the quantity of kilograms of oxidizer, necessary for the complete oxidation 1 kg. of fuel.

Let us introduce the following designations:

$$Б=(2,664C_г+7,937H_г-O_г);$$
$$В=(O_{ок}-2,664C_{ок}-7,937H_{ок}).$$

If for oxidizing 1 kg. the fuel it is required B kg. of oxygen, and in 1 kg. of oxidizer is contained V kg. of oxygen, then for oxidizing 1 kg. of the fuel it is required B kg. of oxygen, but a value which is greater by as many times as the value V is less than one.

Consequently, for the complete oxidation 1 kg. of fuel it is required B/V kg. of oxidizer.

Therefore the coefficient of the stoichiometric relationship/ratio of propellant components is equal to

$$\varkappa_{стех}=\frac{2,664C_г+7,937H_г-O_г}{O_{ок}-2,664C_{ок}-7,937H_{ок}}. \qquad (7.28)$$

The mass portions of nitrogen, which does not participate in the reactions of burning, do not enter into the composition of the obtained formula, but their effect/action lies in the fact that respectively are decreased the mass portions of the elements/cells, which participate in the reactions.

The formula of analogous form is used also for calculating the coefficient $\varkappa_{стех}$ of the fluorine-bearing fuels/propellants, in this

case instead of oxygen the oxidizing element/cell is fluorine, and coefficients in the equations of burning have other values.

Example 3. To determine the coefficient of the stoichiometric relationship/ratio of components of propellant $\varkappa_{стех}$ for fuel/propellant N_2O_4+ aerozine =50.

Solution. We take the composition of oxidizer and fuel from examples 1 and 2 $C_{ок}=0$; $H_{ок}=0$; $O_{ок}=0,696$; $N_{ок}=0,304$; $C_г=0,20$; $H_г=0,13$; $O_г=0$; $N_г=0.67$.

Using equation (7.28), we obtain

$$\varkappa_{стех}=\frac{2,664\cdot 0,20+7,937\cdot 0,13}{0,696}=\frac{1,564}{0,696}=2,248.$$

For the evaluation of that, how the propellant composition and combustion products differs from stoichiometric relationship/ratio, is used the excess oxidant ratio, equal to the ratio of the coefficients of the real and stoichiometric relationship/ratio of propellant components and designated $\alpha_{ок}$:

$$\alpha_{ок}=\frac{\varkappa}{\varkappa_{стех}}. \qquad (7.29)$$

With the excess of fuel in the fuel/propellant the denominator in expression $\varkappa=\dot{m}_{ок}/\dot{m}_г$ is more than in expression $\varkappa_{стех}=\dot{m}_{ок}/\dot{m}_г$; therefore for the present instance $\varkappa<\varkappa_{стех}$ and $\alpha_{ок}<1$.

Page 109.

Consequantly, for the case of the excess of oxidizer in the fuel/propellant

$$\varkappa > \varkappa_{стех} \text{ and } \alpha_{ок} > 1.$$

For the stoichiometric relationship/ratio of propellant components coefficient $\alpha_{ок}$ is equal to one.

In the chamber/camera ZhRD usually provide certain excess of fuel, i.e., $\alpha_{ок} < 1$ ($\alpha_{ок} = 0.7 - 0.9$).

§ 7.5. Mass propellant composition.

If are known the mass portions of chemical elements in the composition of oxidizer and fuel, and also coefficient $\varkappa$, then the mass portion of element/cell in fuel/propellant $g_{э_т}$ is determined from the formula

$$g_{э_т} = \frac{g_{э_г} + \varkappa g_{э_{ок}}}{1+\varkappa}, \tag{7.30}$$

where $g_{э_г}$ and $g_{э_{ок}}$ - mass portions of element/cell in the fuel and the oxidizer respectively.

The construction of formula (7.30) can be confirmed by the following consideration: value $g_{э_г}$ is equal to a quantity to a quantity of element/cell in 1 kg. of fuel, and value $\varkappa g_{э_{ок}}$ - to quantity of element/cell in $\varkappa$ kg. of oxidizer; therefore numerator as a whole is equal to a quantity of element/cell in $(1+\varkappa)$ kg. of

fuel/propellant. Consequently, for obtaining the mass portion of element/cell in 1 kg. of fuel/propellant it is necessary expression $g_{\text{э}_{\text{г}}}+\varkappa g_{\text{э}_{\text{ок}}}$ to divide into value $1+\varkappa$.

On the basis of equation (7.30) calculation of the mass portions of the elements/cells of the fuel/propellant, which contains carbon, hydrogen, oxygen and nitrogen, conduct through the following the formula:

$$\left.\begin{aligned} C_{\text{т}}&=\frac{C_{\text{г}}+\varkappa C_{\text{ок}}}{1+\varkappa}; \quad H_{\text{т}}=\frac{H_{\text{г}}+\varkappa H_{\text{ок}}}{1+\varkappa};\\ O_{\text{т}}&=\frac{O_{\text{г}}+\varkappa O_{\text{ок}}}{1+\varkappa}; \quad N_{\text{т}}=\frac{N_{\text{г}}+\varkappa N_{\text{ок}}}{1+\varkappa}. \end{aligned}\right\} \qquad (7.31)$$

Chemical propellant composition (initial working medium/propellant) can be also expressed in the kilo-atoms of element/cell on 1 kg. of the fuel/propellant:

$$[\mathrm{Э}]_{j\,\text{т}}=\frac{g_{\text{э}_{j\text{т}}}}{A_{\text{э}_j}}, \qquad (7.32)$$

where $[\mathrm{Э}]_{j\text{т}}$ - symbol of the j chemical element, entering the propellant composition.

If are known the total mass propellant component flow $\dot{m}$ and coefficient $\varkappa$, then, by using equation (7.29) and relationship/ratio $\dot{m}=\dot{m}_{\text{ок}}+\dot{m}_{\text{г}}$, it is possible way simple algebraic transformations to obtain the following equations for calculating the values $\dot{m}_{\text{ок}}$ and $\dot{m}_{\text{г}}$:

$$\dot{m}_{\text{ок}}=\frac{\varkappa}{1+\varkappa}\dot{m}; \quad \dot{m}_{\text{г}}=\frac{\dot{m}}{1+\varkappa}. \qquad (7.33)$$

Page 113.

§ 7.6. Generalized system of equations of the determination of the composition of gases.

Let us examine the generalized system of equations, necessary for determining the equilibrium composition of gases in which can flow/occur/last any chemical reactions of dissociation. Such gases can be the products of combustion or decomposition of chemical fuel/propellant or the products of heating working medium/propellant.

Let us introduce the following designations:

m - number of j elements/cells, entering this mixture of gases; n - number of i gases, entering this mixture; z - number of atoms of the j element/cell in the i gas.

Difference m-n is a number of molecular gases, entering this mixture.

For the compilation of the equations of the determination of the composition of the mixture of gases let us write the generalized reaction of the dissociation of the i molecular gas to the atomic gases, i.e., the reaction of atomization which taking into account of

those accepted above designations takes the following form:

$$i\text{-е вещество}^{(1)} \longrightarrow \sum_{j=1}^{j=m} z_j Э_j. \qquad (7.34)$$

Key: (1). substance.

For example, the reaction of the atomization of water vapor takes the form

$$H_2O \longrightarrow 2H + O.$$

Express through the partial pressures of gases, entering the mixture, the generalized equilibrium constant, which corresponds to reaction (7.34):

$$K_i = \frac{\Pi p_{ij}^{z_j}}{p_i}, \qquad (7.35)$$

where Π is the symbol of multiplication of components of the type $p_{ij}^{z_j}$; p_{ij} - the partial pressure of i atomic gas from the j element/cell; p_i - partial pressure of j molecular gas.

For example, the equation of equilibrium constant, which corresponds to the reaction of the atomization of water vapor, takes the form

$$K_{H_2O} = \frac{p_H^2 p_O}{p_{H_2O}}.$$

A number of such equations of equilibrium constants comprises m-n; from a total number of gases, equal m, it is necessary to deduct n of atomic gases which are immune to further dissociation.

Page 111.

In addition to the equations indicated it is possible on the basis of the law of conservation of mass to compose the equations of the conservation of mass of the elements/cells: the mass portions of each element/cell in 1 kg. of initial working medium/propellant and in 1 kg. of the mixture of gases in the chamber/camera are equal to each other.

The generalized equation of the conservation of mass of elements/cells takes the form

$$[\Im]_{j\,\text{т}}=\sum_{i=1}^{i=n} z_j M_i. \qquad (7.36)$$

where $[\Im]_{j\text{т}}$ - number of kilo-atoms of the j element/cell in 1 kg. of initial working medium/propellant, determined according to equation (7.32); M_i - number of kilomoles or kilo-atoms of the i gas, which contains the j element/cell, in 1 kg. of the mixture of gases.

Actually in equation (7.36) a number of products $z_j M_i$ not n, but it is less, since in some i gases the j element/cell is absent (number z_j for them is equal to zero).

A number of equations of the conservation of mass of elements/cells is equal to their number (n).

Express is equation (7.36) through the partial pressures of i gases, which contain the j element/cell, using equation (7.18):

$$[\mathcal{E}]_{j\,\tau}=\frac{M_{\Sigma}}{p_{\Sigma}}\sum_{i=1}^{i=n} a_{ij}p_{i},$$

or taking into account relationship/ratio (7.32)

$$\mathcal{E}_{j\,\tau}=\frac{\mu_{j}M_{\Sigma}}{p_{\Sigma}}\sum_{i=1}^{i=n} a_{ij}p_{i}. \qquad (7.37)$$

In equation (7.37) appear two unknown values - p_{Σ} and M_{Σ}. However, at prescribed/assigned pressure p_{Σ} it is possible to use the equation of Dalton which for the cross sections at the nozzle entry and at output from it takes the following form:

$$\left.\begin{aligned} p_{\Sigma}=p_{\kappa}=\sum_{i=1}^{i=n} p_{i};\\ p_{\Sigma}=p_{c}=\sum_{i=1}^{i=n} p_{i}.\end{aligned}\right| \qquad (7.38)$$

Value M_{Σ} is determined by equation (7.17), but it to it is more convenient consider additional unknown.

Page 112.

Consequently, the generalized system of equations, utilized for determining the composition of mixture m of gases in which can occur any chemical reactions, consists of m+1 equations into number of which they enter:

1) (m-n) the equations of the constants of the chemical equilibrium of the reactions of atomization (7.35);

2) n of the equations of the conservation of mass of elements/cells (7.37);

3) the equation of Dalton (7.38).

A number of unknowns is equal m+2 (m of partial gases, p_{Σ} and M_{Σ}).

The obtained system of equations can be solved in two ways.

1. Values p_{Σ} and M_{Σ} formally recognize by unknowns, and n of equations of conservation of mass of elements/cells (7.37) taking into account equation (7.19) write/record in the following form:

$$g_{\partial_{j}{}_{T}} = \frac{A_{\partial_{j}} \sum_{i=1}^{i=n} z_{j} p_{ij}}{\sum_{i=1}^{i=n} p \mu_{i}}. \qquad (7.39)$$

2. One of equations conservation of mass of elements/cells eliminates by division into it of other equations indicated (in this case it is eliminated also relation M_{Σ}/p_{Σ}). This method widely is used especially with a small number of elements/cells in the working medium/propellant.

The obtained system of equations they solve by successive approximations. With three and more elements/cells in the composition of working medium/propellant the solution of equations is obtained sufficiently to bulky and work consuming ones, in connection with which carry it out in electronic computers [2].

In certain cases, for example for calculation ZhRD, which works on the fuel/propellant of oxygen+hydrogen or fluoro+hydrogen, the system of equations can be solved, also, with the aid of the adding machines.

Is examined below the method of the solutions of system of equations for determining the composition of working medium/propellant based on the example of calculation of composition of combustion products in the chamber/camera of oxygen-hydrogen ZhRD.

§ 7.7. Determination of composition of combustion products based on the example of oxygen-hydrogen fuel/propellant.

During the reaction of oxygen with hydrogen (n=2) are formed the following gases: H_2O, OH, O_2, H_2, O and H, therefore, m=6. Let us write four (m-n=6-2=4) possible ones of the equation of the

equilibrium constants of the reactions of atomization, using equation (7.35):

$$K_{O_2}=\frac{p_O^2}{p_{O_2}}; \qquad (7.40)$$

$$K_{H_2}=\frac{p_H^2}{p_{H_2}}; \qquad (7.41)$$

$$K_{OH}=\frac{p_O p_H}{p_{OH}}; \qquad (7.42)$$

$$K_{H_2O}=\frac{p_H^2 \cdot p_O}{p_{H_2O}}. \qquad (7.43)$$

Page 113.

For the fuel/propellant oxygen + hydrogen mass portions $O_{ok}=1$ and $H_r=1$; therefore the relation mass of the portion of oxygen and hydrogen is equal to x or $\alpha_{ok}x_{ctex}$. Taking into account this and using equation (7.39), let us write the equations of the conservation of elements/cells for oxygen and hydrogen and let us divide their one into another:

$$\alpha_{ok}x_{ctex}=\frac{15,999}{1,008}\cdot\frac{2p_{O_2}+p_O+p_{OH}+p_{H_2O}}{2p_{H_2}+p_H+p_{OH}+2p_{H_2O}}.$$

Taking into account that $\frac{15.999}{1,008}=2x_{ctex}$ (see § 7.4), and introducing value $a=\alpha_{ok}/2$, latter/last equation can be written in the following form:

$$a=\frac{2p_{O_2}+p_O+p_{OH}+p_{H_2O}}{2p_{H_2}+p_H+p_{OH}+2p_{H_2O}}. \qquad (7.44)$$

Let us write the equation of Dalton for the cross section at the nozzle entry

$$p_z=p_k=p_{O_2}+p_O+p_{H_2}+p_H+p_{OH}+p_{H_2O}. \qquad (7.45)$$

Equations (7.40), (7.41), (7.42), (7.43), (7.44) and (7.45) form the system whose solution makes it possible to determine composition of combustion products of oxygen-hydrogen fuel/propellant at the prescribed/assigned values of their temperature and pressure.

Calculation is conducted at a prescribed/assigned (it is more accurate, selected) temperature. The values of equilibrium constants K_{O_2}, K_{H_2}, K_{OH} and K_{H_2O} for the temperature indicated take from handbook [15], where they are given in the physical atmosphere (phys. atm.); therefore prescribed/assigned pressure ($p_\Sigma = p_к$ or $p_\Sigma = p_c$) also must be expressed in the physical atmosphere. for example, if pressure is prescribed/assigned in the bars, then is used the relationship/ratio

$$p\,[\text{phys. atm.}] = \frac{p}{1{,}013}\,[\text{bars}].$$

The obtained system of equations they solve, after assigning pressure p_{H_2}. on the basis of equation (7.41)

$$p_H = \sqrt{K_{H_2} p_{H_2}}.$$

Let us write equations (7.43), (7.42) and (7.40) in the following form:

$$p_{H_2O} = \frac{p_H^2}{K_{H_2O}}\, p_O; \qquad (7.46)$$

$$p_{OH} = \frac{p_H}{K_{OH}}\, p_O; \qquad (7.47)$$

$$p_{O_2} = \frac{1}{K_{O_2}}\, p_O^2. \qquad (7.48)$$

Page 114.

Let us substitute latter/last equations into equality (7.44):

$$a = \frac{\frac{p_H^2}{K_{H_2O}} p_O + \frac{p_H}{K_{OH}} p_O + \frac{2}{K_{O_2}} p_O^2 - p_O}{2p_{H_2} + p_H + \frac{p_H}{K_{OH}} p_O + \frac{2p_H^2}{K_{H_2O}} p_O}.$$

The obtained equation is quadratic equation relatively p_O; the equation indicated it is possible by algebraic transformations to reduce to the following form:

$$\frac{2}{K_{O_2}} p_O^2 + \left[1 - (2a-1)\frac{p_H^2}{K_{H_2O}} - (a-1)\frac{p_H}{K_{OH}}\right] p_O - a(2p_{H_2} + p_H) = 0.$$

Let us designate expression in the brackets by letter A and solve equation relatively p_O:

$$p_O = \frac{A \pm \sqrt{A^2 - \frac{8}{K_{O_2}} a(2p_{H_2} + p_H)}}{\frac{4}{K_{O_2}}}. \qquad (7.49)$$

It is necessary to note that there is only a one system of roots, which satisfies any system of equations in question. for example, if during the solution of equation (7.49) are obtained the positive and negative values of partial pressure, then for calculations we use only a positive value. If both values are positive, then it is necessary to exclude that from them, which with the substitution at least into one of equations leads to the negative value of the partial pressure of any gas.

After determining pressure p_O, we find through equations (7.46), (7.47) and (7.48) pressure p_{H_2O}, p_{OH} and p_{O_2}.

We determine the sum of the partial pressures:

$$p_\Sigma = p_{H_2} + p_H + p_{H_2O} + p_{OH} + p_O + p_{O_2}.$$

If $p_\Sigma > p_\kappa$, then they solve system of equations in the second approximation/approach, in this case are assigned by new, reduced value p'_{H_2}:

$$p'_{H_2} = \frac{p_\kappa}{p_\Sigma} p_{H_2};$$

when $p_\Sigma < p_\kappa$ they are assigned by increased values p'_{H_2}:

$$p'_{H_2} = \frac{p_\kappa}{p_\Sigma} p_{H_2}.$$

Page 115.

It is usually sufficient three, maximum of four, approximations/approaches, in order to with the low error (not more than +-1o/o) was provided the equation

$$p_\Sigma = p_\kappa.$$

The solution of the system indicated is convenient to conduct in the form of the expanded/scanned table (see Table 7.1 V § 7.11; numerals in the brackets of table designate the number of line).

§ 7.8. Determination of the parameters of combustion products at the

nozzle entry.

For determining the basic parameters of chamber/camera it is necessary to design only two values for the entry into its nozzle: the apparent molecular weight of combustion products μ_{κ} (or gas constant R_{κ}) and their entropy s_{κ}. However, for their finding is required to calculate temperature and composition of combustion products on the nozzle entry, and for calculating the latter - total enthalpy of products at three temperatures.

The apparent molecular weight of combustion products is designed from equation (4.6).

Let us examine equations for calculating the entropy of combustion products.

The entropy of individual substances depends on their structure (chemical nature), temperature and pressure [1].

FOOTNOTE [1]. In more detail about the entropy see [18]. ENDFOOTNOTE.

In handbook [15] are given the values of entropy s_0 in cal/(mole•deg) at the standard pressure p_0=1 of phys. atm.; values s_0, undertaken from handbook [15], it is necessary to convert according to the

formula

$$s_0[\text{кдж}^{(1)}/(\text{кмоль}\cdot\text{град})]=4{,}187\,s_0[\text{кал}^{(2)}/(\text{моль}\cdot\text{град})].$$

Key: (1). kJ/(kmole•deg). (2). cal/(mole•deg).

The entropy of individual substance at the arbitrary pressure p, expressed in the phys. atm., is determined according to the equation

$$s_p=s_0-R\ \ln p.$$

After substituting into the latter/last equation value of R=8215 n•m/(kmole•deg)=8315 of kJ/(kmole•deg) and after considering relationship/ratio ln p=2.303 lg p, we will obtain the following formula for calculating the entropy in kJ/(kmole•deg):

$$s_p=s_0-19{,}135\lg p. \qquad (7.50)$$

Page 116.

Entropy the mixtures of gases of different composition, such as are products combustion, design from the formula, analogous (7.21):

$$s_{p_\Sigma}=\frac{\sum_{i=1}^{i=n} s_{p_i} p_i}{\sum_{i=1}^{i=n} \mu_i p_i},$$

which taking into account equation (7.50) can be written in the following final form:

$$s_{p_\Sigma}=\frac{\sum_{i=1}^{i=n}(s_{0i}p-19{,}135p_i\lg p_i)}{\sum_{i=1}^{i=n}\mu_i p_i}. \qquad (7.51)$$

Temperature and composition of combustion products at the nozzle entry determine by utilization equations (7.1), which for the chamber/camera of chemical rocket engines takes the form

$$i_{п.т}=[i_{п.к}]_{T_к}, \qquad (7.52)$$

where $i_{п.т}$ - the total enthalpy of fuel/propellant at the entry into the chamber/camera; $[i_{п.к}]_{T_к}$ - the total enthalpy of combustion products at a temperature $T_к$.

Calculation the temperatures and composition of combustion products at the nozzle entry conduct in the following sequence.

1. They are assigned by temperature $T_к'$ in area of expected value, which can be taken from the thermal calculations which were made earlier, from tables (in particular from Table 10.5 etc.

2. Is designed composition of combustion products at prescribed/assigned temperature employing procedure, examined into § 7.7, or employing other procedures [38].

3. They calculate total enthalpy of fuel/propellant $i_{п.т}$ and combustion products $i_{п.к}'$ at temperature $T_к'$ according to formulas (7.15) and (7.21).

4. Will be brought in obtained value $i_{п.к}'$ for graph $i_{п.к}=f(T_к)$. The

total enthalpies of the combustion products of chemical fuels/propellants have negative values, so that to more conveniently build graph $i_{п.к}=f(T_к)$.

5. Compare obtained value $i'_{п.к}$ with total enthalpy fuels/propellants $i_{п.т}$. If absolute value $i'_{п.к}$ more than absolute value $i_{п.т}$, then it is necessary to assign greater temperature $T_к$ and vice versa. For the construction of more accurate graph $i_{п.к}=f(T_к)$ they are assigned by three values of temperature $T'_к$ and are designed composition and values $i_{п.к}$ for each of its values.

Page 117.

6. Build graph $i_{п.к}=f(T_к)$ (Fig. 7.1) and find through it temperature of products combustion on nozzle entry $T_к$, using relationship/ratio (7.52).

7. Define classification of products combustion at obtained temperature, for which on three points ($T'_к$, $T''_к$ and $T'''_к$) are built graphs $p_i=f(T_к)$. Graphs it is convenient to build by those combined (on one sheet), selecting for each partial pressure such scale so as as far as possible to use entire field of sheet. Fig. 7.2 shows finding partial pressures p_{H_2O} and p_{OH} graphs $p_{H_2O}=f(T_к)$ and $p_{OH}=f(T_к)$.

8. Determine apparent molecular weight of products combustion $\mu_{\Sigma к}$ at obtained temperature, for which is built graph $\mu_{\Sigma к}=f(T_к)$. Values $\mu'_{\Sigma к}, \mu''_{\Sigma к}$ and $\mu'''_{\Sigma к}$ for temperatures $T'_к, T''_к$ and $T'''_к$ design from equation (4.6).

9. Find through equation (4.5) gas constant of products combustion on nozzle entry.

10. Determine according to equation (7.51) entropy of products combustion at temperatures $T'_к, T''_к$ and $T'''_к$ and graphically at temperature $T_к$ is found entropy $s_к$ (see Fig. 7.1).

Calculation examined above is conveniently conducted, filling for temperatures $T'_к, T''_к, T'''_к$ and obtained temperature $T_к$ table 7.2 (see § 7.11).

The sums of calculation at temperatures $T'_к, T''_к$ and $T'''_к$ and also the results of graphic construction will bring in in table 7.3 (see § 7.11).

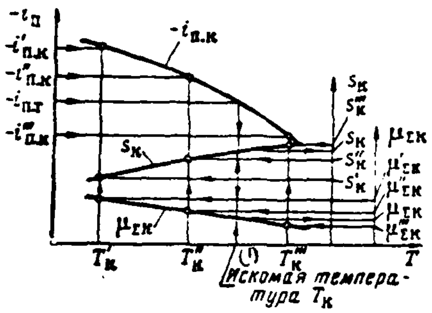


Fig. 7.1.

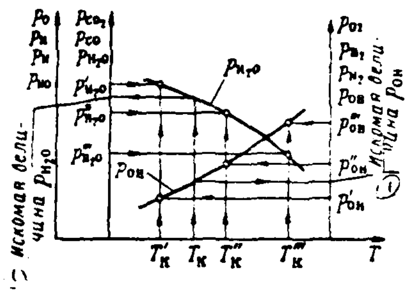


Fig. 7.2

Fig. 7.11. Graphs for determining the values $T_к$, $s_к$ and $\mu_{\Sigma к}$ from known value $i_{п.т}$ (according to the results of calculation at three selected values $T_к'$, $T_к''$ and $T_к'''$)

Key: (1). Unknown temperature.

Fig. 7.2. Graphs for determining composition of combustion products at obtained temperature $T_к$ (according to results of calculation at three selected values $T_к'$, $T_к''$ and $T_к'''$)

Key: (1). Unknown value.

Page 118.

§7.9. Determination of the parameters of combustion products at the nozzle outlet.

Calculation of the parameters of combustion products in nozzle exit section in many respects is analogous to calculation, examined in §7.8. Differences consist in the fact that instead of equation (7.52) is used equation (7.2) and instead of condition $p_{к}=\sum_{i=1}^{i=n} p_i$ is provided condition $p_c=\sum_{i=1}^{i=n} p_i$.

Due to a comparatively low temperature of combustion products at the nozzle outlet some partial pressures (for example, p_O and p_H) are very low and it is possible not to consider them during calculation of the parameters in this cross section.

Calculation of the parameters into the outlet cross section nozzles conduct in the following sequence.

1. Given by tentative value of index of equilibrium process of expansion n_p (e.g., from Table 10.5), is determined on formula (4.22)

expected temperature of combustion products at nozzle outlet (we accept $R_к = R_c$)

$$T_c \approx T_к \left(\frac{p_c}{p_к}\right)^{(n_p-1)/n_p}. \qquad (7.53)$$

2. They are assigned by three values of temperature T_c', T_c'' and T_c''' in area of expected temperature, and for each value T_c is designed composition of products of combustion.

3. Find values of entropy s_c', s_c'' and s_c''' for each value of temperatures.

4. Is built graph $s_c = f(T_c)$ (Fig. 7.3), and from condition $s_к = s_c$ are determined temperatures of products combustion in nozzle exit section.

5. Define classification of products combustion at obtained temperature T_c, using graphs $p_i = f(T_c)$, which can be constructed by obtained composition of combustion products at temperatures T_c', T_c'' and T_c'''.

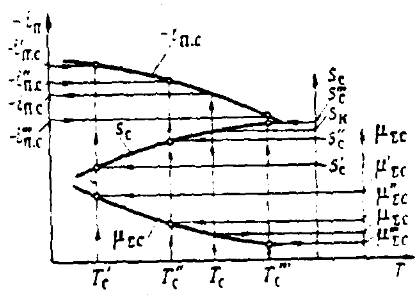


Fig. 7.3. Graphs for determining the values T_c, $i_{\pi.c}$ and $\mu_{\Sigma c}$ from known value $s_c = s_K$ (according to the results of calculation at three selected values T_c', T_c'' and T_c''').

Page 119.

6. By analogy with calculation, examined in §7.8, determine apparent molecular weight $\mu_{\Sigma c}$ and gas constant $R_{\Sigma c}$ of products combustion in nozzle exit section.

7. As during calculation in §7.8, calculations conduct, filling for each value of temperature T_c Tables 7.2 and 7.3 (see §7.11).

§7.10. Calculation of specific impulse and sizes/dimensions of nozzle taking into account losses.

In calculations, examined in the preceding paragraphs, are considered only the phenomena, connected with the dissociation and

the recombination, and the losses, connected with the carry-off of heat and chemical energy together with the products of combustion behind the nozzle. Formula for determining the solidity ratio of specific impulse at the level of sea, which considers remaining losses, can be taking into account equations (4.40) and (4.41) written in the following form:

$$\varphi_I=\frac{I_{\text{уд.з.д}}}{I_{\text{уд.з}\,t}}. \qquad (7.54)$$

Specific impulse $I_{\text{уд.з.д}}$ is designed from equation (7.23), the values of thrust $P_з$ and expenditure of components of fuel $\dot m=\dot m_{\text{ок}}+\dot m_{\text{г}}$ measuring during the bench test of engine. The order of calculation of value $I_{\text{уд.з}t}$ is examined in §7.1.

If we during engine testing ensure conditions $p_{\text{к.д}}=p_{\text{к}t}$, $\dot m_д=\dot m_t$ and $x_д=x_t$, then equation (4.37) for calculating the coefficient φ_β can be taking into account formula (4.14) written in the form

$$\varphi_\beta=\frac{f_{\text{кр.д}}}{f_{\text{кр}\,t}}, \qquad (7.55)$$

where $f_{\text{кр.д}}$ - throat area of real chamber/camera; $f_{\text{кр}t}$ - throat area of the chamber/camera for which $\varphi_I=1$; this area is calculated from equation (4.14).

In terms of the values of coefficients φ_I and φ_β, obtained from equations (7.54) and (7.55), it is possible, using formula (4.41), to find coefficient φ_c:

$$\varphi_c = \frac{\varphi_l}{\varphi_\beta}. \tag{7.56}$$

If calculation of chamber/camera is performed before the bench tests of engine on results of which can be determined coefficients φ_β and φ_c, then their value is selected, using results of the tests of the chambers/cameras, which have analogous construction/design and which work on the same fuel/propellant.

Page 120.

After selecting thus coefficients φ_β and φ_c, is designed from equation (4.40) value $I_{\text{уд.д}}$ and from formula (1.23) the actually/really necessary fuel consumption:

$$\dot{m}_{д} = \frac{P_з}{I_{\text{уд.з.д}}}. \tag{7.57}$$

If we write equation (7.57) for case $\varphi_l = 1$ and divide the first equation into that obtained, and also consider condition $P_{з.д} = P_{зt}$ and equation (4.40), then

$$\frac{\dot{m}_д}{\dot{m}_t} = \frac{1}{\varphi_\beta \varphi_c}. \tag{7.58}$$

If we for the real chamber/camera hold out condition $p_{к.д} = p_{кt}$, then in accordance with equations (4.14) and (4.37) the area of its critical cross section can be determined according to the equation

$$\frac{f_{\text{кр.д}}}{f_{\text{кр}\,t}} = \varphi_\beta \frac{\dot{m}_д}{\dot{m}_t},$$

which taking into account equation (7.58) is possible to write in the simpler form:

$$f_{\text{кр д}} = \frac{f_{\text{кр}t}}{\varphi_c}. \qquad (7.59)$$

If we consider that index n_p for the real chamber/camera and for the chamber/camera whose coefficient φ_t is equal to one, is identical, is provided equality $\bar{f}_{c.д} = \bar{f}_{ct}$, and for calculating the area of nozzle exit section of real chamber/camera it is possible to use the equation

$$f_{c.д} = \frac{f_{ct}}{\varphi_c}. \qquad (7.60)$$

§7.11. Example of the thermal design ZhRD, operating on the oxygen-hydrogen fuel/propellant.

To conduct the thermal design of liquid propellant rocket engine for the following prescribed/assigned parameters.

1. Empty thrust $P_п = 100$ кп.

2. Fuel/propellant: liquid oxygen + liquid hydrogen; coefficient $\alpha_{ок} = 0{,}8$.

3. Pressure of combustion products at nozzle inlet $p_к = 100$ bar.

4. Pressure of combustion products in nozzle exit section $p_c = 0{,}5$ bar.

5. Coefficients $\varphi_{\beta}=0{,}98$; $\varphi_{c}=0{,}99$.

Solution. 1. Let us extract known data for fuel/propellant liquid oxygen + liquid hydrogen:

a) coefficient $\varkappa_{стех}=7{,}937$ (see §7.4);

b) total enthalpy of liquid oxygen at boiling point $i_{п.ок}=-398$ kJ/kg (see Table 10.3);

c) complete enthalpy of liquid hydrogen at boiling point $i_{п.г}=-3828$ kJ/kg (see Table 10.4).

Page 121.

2. We find through equation (7.29) the actual value of coefficient $\varkappa$:

$$\varkappa=\alpha_{ок}\varkappa_{стех}=0{,}8\cdot 7{,}937=6{,}350.$$

3. We determine according to equation (7.15) total enthalpy of fuel/propellant at the inlet into chamber/camera (we consider that temperature of propellant components is equal to their boiling point):

$$i_{п.т} = \frac{i_{п.г} + \varkappa i_{п.ок}}{1 + \varkappa} = \frac{-3828 + 6,350\,(-398)}{1 + 6,350} = -864,7 \text{ kJ/kg.}$$

Calculation of the parameters of combustion products at the nozzle entry.

4. In accordance with Table 10.5 temperature $T_к$ of combustion products of oxygen-hydrogen fuel comprises when $\alpha_{ок}=0,505$ $3000°K$; for $\alpha_{ок}=0,8$ it has higher value. We select $T_к=3600°K$.

5. We express pressure $p_к$ in physical atmosphere:

$$p_к = \frac{100}{1,013} = 98,716 \text{ phys. atm.}$$

6. We determine value a in equation (7.44):

$$a = \frac{\alpha_{ок}}{2} = \frac{0,8}{2} = 0,4.$$

7. We will calculate composition of combustion products at temperature of 3600°K, using system of equations, examined in §7.7.

Calculation we conduct according to Table 7.1, in which before beginning calculation we fill lines from the first to the sixth (values of equilibrium constants K_{H_2}, K_{O_2}, K_{H_2O} and K_{OH} we take from the handbook [15] for temperature 3600°K), and also lines 10-12

(a=0.4). In the first approximation, we are assigned by the partial pressure of molecular hydrogen $p_{H_2}=9$ phys. atm.

8. According to results of calculation we fill column 3 Table 7.2.

9. We determine according to equation (4.6) apparent molecular weight of combustion products at nozzle inlet (see column 4 of Table 2):

$$\mu_{\text{г}\kappa}=\frac{1}{p_{\kappa}}\sum_{i=1}^{i=n}\mu_i p_i=\frac{1379,3}{98,45}=14,010 \text{ kg/kmole.}$$

10. We will calculate according to equation (7.21) total enthalpy of combustion products at nozzle inlet. Values $i_{n\,i}$ (column 5, Table 7.2) we take from handbook [15] in cal/mole and shift in kJ/kmole according to the equation

$$i_{n\,i}\,[\text{кдж/кмоль}]^{(1)} = 4,187 i_{n\,i}\,[\text{кал/моль}]^{(2)};$$

$$i_{n.\kappa}=\frac{\sum\limits_{i=1}^{i=n} i_{n\,i}p_i}{\sum\limits_{i=1}^{i=n}\mu_i p_i}=\frac{-994421,2}{1379,3}=-721,0 \text{ кдж/кг}^{(3)}.$$

Key: (1). kJ/kmole. (2). cal/mole. (3). kJ/kg.

Page 122. Table 7.1. Calculation of composition of combustion products of oxygen-hydrogen fuel/propellant with $T_к=3600$ K, $p_к=98{,}716$ the phys. atm. and $\alpha_{ок}=0{,}8$

(1) № строки	(2) Величина	(3) Численное значение величины (4) Приближение: первое	второе	третье
1	$T_к$		3600	
2	$p_к$		98,716	
3	K_{H_2}		$5{,}365\cdot10^{-1}$	
4	K_{O_2}		$3{,}916\cdot10^{-1}$	
5	K_{H_2O}		$7{,}956\cdot10^{-2}$	
6	K_{OH}		$3{,}053\cdot10^{-1}$	
7	p_{H_2}	9	22,21	21,17
8	$[3]\cdot[7]^2$	4,828	11,915	11,357
9	$p_H=\sqrt{[8]}$	2,197	3,452	3,370
10	a		0,4	
11	$2a-1$		—0,2	
12	$a-1$		—0,6	
13	$[8]:[5]$	60,683	149,76	142,74
14	$[11]\cdot[13]$	—12,138	—29,952	—28,548
15	$[9]:[6]$	7,196	—11,306	11,038
16	$[12]\cdot[15]$	—4,317	—6,784	—6,623
17	$1-[14]-[16]$	17,455	37,736	36,171
18	$2[7]+[9]$	20,197	47,872	45,710
19	$[18]\cdot[10]$	8,078	19,148	18,284
20	$[17]^2$	304,677	1424,0	1308,3
21	$[19]:[4]$	20,628	48,896	46,690
22	$8\cdot[21]$	166,025	391,17	373,50
23	$[20]+[22]$	469,702	1815,17	1681,8
24	$\sqrt{[23]}$	21,672	42,61	41,01
25	$-[17]+[24]$	4,217	4,874	4,839
26	$4:[4]$	10,214	10,214	10,214
27	$p_O=[25]:[26]$	0,412	0,4770	0,4738
28	$[27]^2$	0,170	0,2270	0,2245
29	$p_{O_2}=[28]:[4]$	0,434	0,5797	0,5733
30	$p_{H_2O}=[13]\cdot[27]$	25,001	71,435	67,630
31	$p_{OH}=[15]\cdot[27]$	2,964	5,393	5,2298
32	$p_\Sigma=[7]+[9]+[27]+[29]+[30]+[31]$	40,008	103,55	98,45

Key: (1). line. (2). Value. (3). Numerical value of quantity. (4).

Approximation/approach. (5). the first. (6). the second. (7). the third.

FOOTNOTE 1. Numerals in the brackets designate the number of the line from which must be taken the numerical value of quantity. ENDFOOTNOTE.

Page 123.

11. We find through equation (7.51) entropy of combustion products on the inlet into nozzle. Values s_{0i} (column 7 tables 7.2) we take from handbook [15] in cal/(mole•deg) and shift in kJ/(kmole•deg):

$$s_{p\Sigma}=\frac{1}{\sum\limits_{i=1}^{i=n}\mu_i p_i}\sum_{i=1}^{i=n}(s_{0i}p_i-19,135p_i\lg p_i)=$$

$$=\frac{1}{1379,3}\cdot 23787,5=17,246 \overset{(1)}{\text{кдж (кг·град)}}.$$

Key: (1). kJ/(kg•deg).

Table 7.2. Calculation of the parameters of the combustion products of oxygen-hydrogen fuel/propellant at $T_к=3600° K$, $p_к=98,716$ the phys. atm. and $\alpha_{ок}=0,8$

(1) Газ	μ_i (2) кг/кмоль	p_i (3) физ. атм	[2]·[3][1]	i_i (4) кдж/кмоль	[3]·[5]	s_i (5) кдж/кмоль·град
H_2	2,016	21,17	42,6787	111 612,8	2 362 842,98	209,8054
H	1,008	3,37	3,397	286 918,3	966 914,67	166,5107
O	15,999	0,4738	7,5803	318 680,9	150 991,01	213,5872
O_2	31,999	0,5733	18,3450	122 666,5	70 324,70	292,0068
H_2O	18,015	67,630	1218,3545	—78 929,1	—5 337 975,0	297,8254
OH	17,007	5,2298	88,9432	151 531,7	792 480,48	263,8417
Сумма	—	98,45	1379,3	—	—994 421,2	—

(6) Продолжение

(1) Газ	[3]·[7]	lg p_i	19,135 [9]	[3]·[10]	[8]—[11]
H_2	4441,664	1,3257	25,3673	537,0257	3904,6183
H	561,141	0,5276	10,0956	34,0222	527,1188
O	101,198	—0,3244	—6,2074	—2,9411	104,1391
O_2	167,407	—0,2416	—4,6230	—2,6504	170,0574
H_2O	20141,932	1,8301	35,0190	2368,335	17773,597
OH	1379,839	0,7185	13,7485	71,9019	1307,9371
Сумма	—	—	—	—	23787,5

Key: (1). Gas. (2). kg/kmole. (3). phys. atm. (4). kJ/kmole. (5). kJ/kmole•deg. (6). Continuation. (7). Sum.

FOOTNOTE [1]. Numerals in the brackets indicate the number of the column from which must be undertaken the numerical value of quantity.

ENDFOOTNOTE.

Page 124.

12. We conduct analogous calculations for temperatures of 3400 and 3500°K and obtained results bring in in Table 7.3. Using the graphs (see §7.8), we find the value $T_{\kappa}, \mu_{\Sigma\kappa}, i_{п.\kappa}$ and s_{κ}, which also bring in Table 7.3.

13. On the basis of Table 10.5 we select index n_p for equilibrium expansion of combustion products of oxygen-hydrogen fuel and on equation (7.53) it is determined their expected temperature in nozzle exit section:

$$T_c \approx T_\kappa \left(\frac{p_c}{p_\kappa}\right)^{(n_p-1)/n_p} = 3600 \left(\frac{0.5}{100}\right)^{0,232/1,232} = 1330°K.$$

Since the value of index in Table 10.5 for $\alpha_{o\kappa}=0{,}505$, then for calculating the parameters of combustion products at the nozzle outlet when $\alpha_{o\kappa}=0.8$ we are given assigned by the higher values of temperature $T_c=1900, 2000$ and 2100°K.

14. We conduct calculation of composition of combustion products at temperatures indicated and pressure $p_c=\frac{0.5}{1{,}013}=0{,}494$ phys. atm., using Table 7.1.

The values of equilibrium constants we take from handbook [15].

15. After determining composition of combustion products with T_c=1900, 2000 and 2100°K, we fill for each of them table analogously to Table 7.2.

16. Sums of calculation bring in Table 7.4.

Table 7.3. Results of calculating the parameters of combustion products at the nozzle inlet (at three prescribed/assigned and obtained values of temperature $T_к$)

$T_к$ °K	$\mu_{\Sigma к}$ (1) $\frac{кг}{кмоль}$	$i_{п.к}$ (2) $\frac{кдж}{кг}$	$s_к$ (3) $\frac{кдж}{кг \cdot град}$
3400	14,33	—[illegible]	[illegible]
3500	14,19	—[illegible]	[illegible]
3600	14,01	—721,9	17,25
3580	14,04	—864,6	17,20

Key: (1). kg/kmole. (2). kJ/kg. (3). kJ/kg•deg.

Table 7.4. Results of calculating the parameters of combustion products at the nozzle outlet (at three prescribed/assigned and obtained values of temperature T_c)

T_c °K	$\mu_{\Sigma c}$ (1) $\frac{кг}{кмоль}$	$R_{\Sigma c}$ (2) $\frac{н \cdot м}{кг \cdot град}$	s_c (3) $\frac{кдж}{кг \cdot град}$	$i_{п.c}$ (4) $\frac{кдж}{кг}$
1900	14,80	562,2	17,33	—8677,1
2000	14,78	562,4	17,54	—8351,4
2100	14,77	563,2	17,79	—7997,1
1950	14,79	562,3	17,33	—8499,6

Key: (1). kg/kmole. (2). n•m/kg•deg. (3). kJ/kg•deg. (4). kJ/kg.

Page 125.

16. Using data of Tables 7.3 and 7.4 and equations, examined in §§7.1 and 7.10, we find basic parameters of chamber/camera. Their calculation is conducted, filling Table 7.5.

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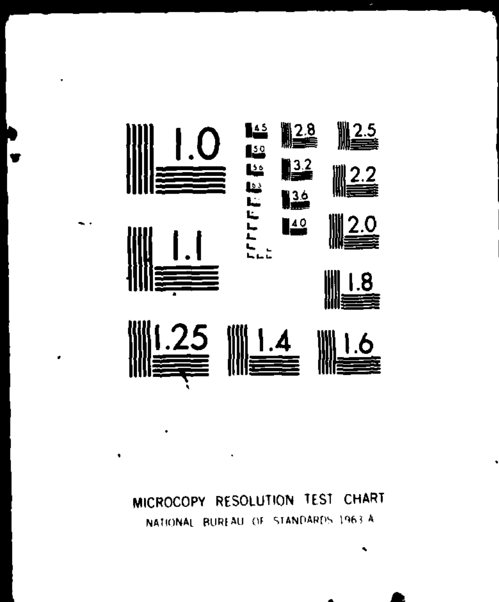

Table 7.5. Calculation of the basic parameters of chamber/camera.

(1) Обозначение	(2) Размерность	(3) Формула для определения	(4) Численное значение	(1) Обозначение	(2) Размерность	(3) Формула для определения	(4) Численное значение
W_c	$\frac{м}{сек}$	(7.3)	3908	$I_{уд.п.д}$	$\frac{н \cdot сек}{кг}$	(4.40)	4063
ϱ_c	$\frac{кг}{м^3}$	(7.4)	0,0456	$\dot{m}_д$	$\frac{кг}{сек}$	(1.17)	24,61
$I_{уд.п\,t}$	$\frac{н \cdot сек}{кг}$	(7.5)	4188	$\dot{m}_{ок}$	$\frac{кг}{сек}$	(7.33)	21,26
$\dot{m}_t$	$\frac{кг}{сек}$	(7.6)	23,88	$\dot{m}_г$	$\frac{кг}{сек}$	(7.33)	3,35
$f_{c\,t}$	$м^2$	(7.7)	0,1340	$f_{кр.д}$	$м^2$	(7.59)	0,00551
n_p	—	(4.7) или (4.8)	1,142	$f_{с.д}$	$м^2$	(7.60)	0,135
				$P_{в.д}$	$н$	(1.12)	86 500
β_t	$\frac{н \cdot сек}{кг}$	(7.8)	2284,7	$I_{уд.в.д}$	$\frac{н \cdot сек}{кг}$	(1.23)	3515
$f_{кр\,t}$	$м^2$	(7.9)	0,005456				

Key: (1). Designation. (2). Dimension. (3). Formula for determining. (4). Numerical value. (5). s. (6). N•s. (7). kg. (8). or.

Chapter VIII.

ENGINE INSTALLATIONS WITH THE THERMAL ROCKET ENGINES.

§8.1. Engine installations of rocket vehicles.

Should be distinguished concepts "engine" and "engine installation" (DU).

Engine are called chamber/camera and totality of aggregates, assemblies and conduits/manifolds, which accomplish/realize supply of working medium/propellant (propellant components) into the chamber/camera, and in a number of cases and creating efforts/forces and moments for the control of rocket vehicle.

Page 126.

The units indicated and aggregates place directly on the chamber/camera or on the special frame, utilized for fastening of engine to the power ring (frame/former) of rocket vehicle. Through the frame is transmitted to the frame/former the thrust, developed with engine.

Liquid working medium/propellants are supplied the chamber/camera, which has the relatively large thrust, with the pumps, entering together with the turbine into the so-called turbopump aggregate (TNA).

For the work of turbine, utilized for the pump drive, it is necessary to have in the composition of engine an aggregate, which produces gaseous working body with those required by pressure and temperature. As this aggregate can serve special gas generator. Furthermore, gaseous working body can be obtained in the coolant passage of chamber/camera.

Most frequently are used the chemical gas generators, which use liquid or solid propellant components. The gas generator, which works on the liquid propellant, is called the liquid-gas generator (ZhGG), on the solid - by solid-propellant gas generator (TGG). Solid fuel is placed directly in TGG.

The supply of working medium/propellant into aggregates and units of engine is realized by the special system, into which are included different valves, which are opened/disclosed or which are closed at the required moment of time.

In chemical RD in a number of cases proves to be necessary a

system, which ensures the beginning of the reaction of burning or decomposition.

Engines with turbopump assemblies are started via turbine boost by the gas, obtained in the starting/launching gas generator (starter).

In the main of some engines with TNA build in the aggregates of pressurized system, which produce gas which is supplied into the gas cavity of tank with the liquid working medium/propellant and is created in it the pressure, necessary for the normal work of pump.

With starting/launching and engine cutoff in a number of cases it is necessary to blow the cavities of some aggregates, for example combustion chamber. For this purpose the engines equip with tank/balloon with compressed gas (usually inert) and corresponding valves and conduits/manifolds.

For the creation of control forces and moments/torques engine chamber establish/install to the rocket vehicle on by hinge joint or gimbal suspension, change the expenditure of gaseous the working the bodies in front of exhaust nozzles of turbine, etc.

Engine installation encompasses the following aggregates and

systems.

1. Engine. For accomplishing one and the same mission objective in DU of rocket vehicle it is possible to use a mono- or multichamber (usually a number of chambers/cameras is equal to 2-4) engine of the required thrust, and also several single-chamber engines with the same gross thrust.

Page 127.

For the multichamber engine is characteristic the supply of working medium/propellants into all its chambers/cameras by one and the same turbopump unit.

In DU rocket vehicle together with the march ones can be included the pilot engines with the relatively low thrust (helmsmen, brake, etc.).

2. Tanks with working medium/propellants. Earlier (chapter III) it was indicated that for rocket thrust-chamber firing together with the basic working medium/propellant there are necessary the additional and starting/launching working medium/propellants. Within or outside the tanks can be established/installed different aggregates and units: valves, conduits/manifolds of servicing and

drain of working medium/propellants, the conduits/manifolds of their delivery to the engine, etc.

3. Aggregates of pressure feed system. They are necessary in such a case, when in the engine there is no TNA. For example, working body can be displaced into the chamber/camera by the compressed gas which enters tank from the special tank/balloon.

4. Aggregates of pressurized systems, blasting of tanks, if they are not included in engine.

The design features of engine installation and engine are connected with each other. For example, if in DU of rocket vehicle are vernier engines, then drops off the necessity for the hinged or gimbal suspension of main engines. If for the inflating of tanks are used ZhGG, adjusted on the upper bottoms of tanks, then from the engine eliminate the aggregates of pressurized system, built in in its main.

In the engine installations of RDTT engine it is difficult to separate. ZhRD with the pressururized-propellant feed it is possible to establish/install directly within the tank and to weld with it. In this case the tank and engine in structural/design sense are unit.

In DU of rocket vehicle can be included the engines of different types. For example, retro-engine installation of space vehicle "Surveyor" (USA), intended for its soft landing on the moon, is RDTT and three steering ZhRD with controllable thrust.

58.2. Classification of thermal rocket engines on the design features.

Depending on method the supplies of working medium/propellant into the chamber/camera distinguish rocket engines with the pressurization, the pump, etc. systems of supply.

In thermal RD with the prassururizad-propellant feed is working the body is removed from the tank by the gas, previously stored up under the large pressure in the so-called gas storage tank of pressure (in abbreviated form AD) or by the gas, generated in TGG or ZhGG.

The gas, which enters in bais from AD gas, can be preheated by the fact or another method, for example, by the products of combustion of TGG.

Page 128.

The gas, which is generated in ZhGG and used for the pressururized-propellant feed, can be:

1) the decomposition products of additional liquid propellant component;

2) the reaction products of two additional liquid propellant components; to avoid the high temperatures of the products indicated is used significant excess of one of the propellant components, i.e., $\alpha_{ok} \gg 1$ or $\alpha_{ok} \ll 1$.

For the supply of the liquids indicated, which are stored in the separate small tanks, is necessary special aggregate, for example TGG.

In thermal RD with the pump feed liquid working medium/propellants are supplied in barrels of TNA. Such engines distinguish by the method obtainings of the gaseous working medium/propellant, utilized for the feed/supply of turbine TNA. As the gaseous working medium/propellant are used:

1) the decomposition products of liquid basic or additional propellant component in the liquid-gas generator, which in this case is called one-component; for supplying the additional component in

ZhGG is necessary a special system;

2) the reaction products of two basic propellant components in ZhGG moreover so that would not be fused the blades of turbine also necessary to provide condition $\alpha_{ox} \gg 1$ or $\alpha_{ox} \ll 1$; such ZhGG, widely utilized in ZhRD, call two-component;

3) basic are working body, and also basic or one of the basic components of propellant (for example, hydrogen), selected/taken from the cooled channel of chamber/camera);

4) the products of heating working medium/propellant and decomposition products or the reactions of basic propellant components, selected/taken from the basic chamber/camera. Before the supply into the turbine must be cooled them in any manner (for example, by mixing length with the cold working medium/propellant).

Gaseous working body after operation in the turbine can be thrown out through the special nozzles of exhaust pipe into the environment or head in chambers/cameras. In the first case of the nozzle of exhaust pipe they develop augmented thrust.

The gaseous is working body, which enters from the turbine the chamber/camera chemical RD, it can be:

a) one of the basic propellant components, gasified in the coolant passage of the chamber/camera before the supply into the turbine;

b) the decomposition products of one of the basic propellant components in single-/mono-component ZhGG;

c) the reaction products of two basic propellant components in two-component ZhGG.

Liquid working medium/propellants can be supplied with the aid of the ejectors (see §13.1); for this purpose it is possible to use centrifugal forces (if the rocket vehicle it is rotated with the sufficiently large angular velocity), acceleration of rocket vehicle, gravitational field of the Earth or another planet, etc.

Page 129.

Feed systems with the utilization of centrifugal forces were used, in particular, in the engine installations of artificial Earth satellites, stabilized by rotation.

The construction/design of chamber/camera thermal RD to a considerable degree depends on the method of its cooling. Frequently is used external flowing cooling. In this case over the coolant passage, formed by space between the double chamber walls, flows/occurs/lasts working the body, which receives heat fluxes from heating products, combustion or expansions into the chamber walls.

The chambers/cameras whose cooling is provided by other methods, are more simple by the construction/design: in them there is no coolant passage, and therefore there are no inherent in it hydraulic losses. However, at a high temperature of gas and with the prolonged work of engine the creation of such chambers/cameras causes great difficulties.

Chemical RD subdivide in a quantity of propellant components into the the mono-, two- and three-component ones; each component is supplied into the chamber/camera on the separate main.

Page mono- and two-component chemical engines (RDTT, ZhRD) extensively are used in the contemporary rocket vehicles. The promising engines include three-component ZhRD and RDGT, capable of developing highest specific impulse of all types of chemical engines.

In the value of developed rocket engine thrust can be subdivided

as follows:

a) micromotors 1 mn [~~0.1G] - 10 n [~~1000 G];

b) low-thrust engines: 10 n [~~1 kgf] - 10 kn [~~1 T];

c) the engines of average thrust: 10 kn [~~1 T] - 1 mN [~~100 T];

d) the engines of the large thrust: 1 mN [~~100 T] - 10 mN [~~1000 T];

e) the engines of the ultrahigh thrust: it is more than 10 mN [~~1000 T].

Engine installations with the thermal rocket engines can be serviced previously at the same plant, at which is produced the assembly of DU or rocket vehicle as a whole. Rocket vehicle is transported in the charged/filled state and can if necessary to allow/assume prolonged storage and to provide rapid starting/launching after obtaining of command/crew. For example, they previously load by fuel/propellant RDTT, and also series/number of DU with the liquid-propellant engines.

Tanks of DU of space rocket vehicles (especially DU with ZhRD of large thrust) service directly on the launcher. Rocket vehicles with the empty tanks it is sufficiently simple and it is safe to transport, but the places of their launching/starting must be equipped by capacities with the reserve of working medium/propellants and devices for servicing of tanks, which substantially complicates the launcher.

Page 130.

In the number of inclusions/connections thermal RD they subdivide into the engines of one-time and multiplying. The engines of one-time inclusion/connection are used for first stages of rocket-carriers and for the overwhelming majority of single-stage rockets. The engines of multiplying have a more compound circuit and a construction/design, but their utilization makes it possible to noticeably improve the characteristics of rocket vehicle. For example, the carrier rocket, at latter/last step/stage of which is established/installed ZhRD with the reclosing, can derive in orbit the satellite of greater mass than RDTT, which usually connects one time. Are especially necessary engines with multiplying for the space vehicles.

By the character of work distinguish continuous engines and

pulse engines, i.e., the engines, which work in the pulse, or the relay, mode/conditions.

Continuous engines are engines with one-time or multiplying, whose time of continuous operation is considerably more than the time of the output/yield to the nominal rating and of the decay time in the thrust.

For the pulse engines short period of work follows also the short period, during which the engine is switched off, the periods indicated continuously replacing one another. Pulse engines are necessary, in particular, for the stabilization systems and orientation of satellites and space vehicles.

Thermal rocket engines can be subdivided also over possibility and range of a change in the expenditure of working medium/propellant and, consequently, also thrust.

Some engines do not have special aggregates for changing the expenditure of working medium/propellant. Such engines work with the approximately/exemplarily constant expenditure for which them they previously adjust.

The construction/design of the majority of rocket engines

provides for a change of the expenditure of working medium/propellant in the relatively small range (for example, +-5-10o/o of the nominal value).

Two types of engines indicated are used in essence in the single-stage rockets and the first booster stages.

Some rocket engines (for example, braking ZhRD of the space vehicle, intended for the soft landing on the moon or another planet) provide a large (in relation 10:1 and more) reduction/descent in the expenditure of working medium/propellant in comparison with the nominal value.

Page 131.

Part III.

CONSTRUCTION AND DESIGN OF CHEMICAL ROCKET ENGINES.

Chapter IX.

TYPICAL SYSTEMS OF LIQUID PROPELLANT ROCKET ENGINES.

§9.1. Special features/peculiarities of systems of ZhRD [liquid propellant rocket engine].

ZhRD with pressure feed system into the chamber/camera can be subdivided using the method of obtaining the displacing gas on engines with the gas storage tank of pressure, with ZhGG and TGG (see Chapter VIII). The simplest diagram of one of such ZhRD is examined in §1.2; in detail they are described in chapter XIII.

ZhRD with the pump feed system classify according to the state of aggregation of the propellant components, which enter the

chamber/camera, and according to the special features/peculiarities of the branch/removal of working medium/propellant after its operation in the turbine; frequently working body is generator gas, i.e., it is produced in the gas generator.

As it will be shown below, engine it is desirable to project/design so that it would work without the use/application of additional propellant components. Therefore subsequently are examined only such diagrams of ZhRD whose turbine works on the gas, obtained of one or two basic propellant components. Usually chamber/camera is cooled by fuel; this is taken into consideration in all diagrams, examined/considered in present chapter.

ZhRD with the throw-out of the exhaust generator gas into the environment (Fig. 9.1). Oxidizer and combustible are introduced inside the combustion chamber of such ZhRD in the liquid state, i.e., engine works on the diagram "liquid-to-liquid", and the exhaust generator gas is thrown out through the nozzle of turbine exhaust into the environment. The throw-out of the gas indicated decreases specific jet firing. Although the nozzle of turbine exhaust, as already mentioned above, and develops certain thrust, its specific impulse due to the low temperature of generator gas and small degree of its expansion is comparatively low.

Page 132.

In the diagram of ZhRD in question generator gas is the products of the incomplete combustion of two-component fuel/propellant, which contain the large excess of oxidizer ($\alpha_{ok} \gg 1$) or fuel ($\alpha_{ok} \ll 1$). ZhGG, which works when $\alpha_{ok} \gg 1$, they call oxidative, and when $\alpha_{ok} \ll 1$ - reducing.

ZhRD with the supply of exhaust generator gas into the combustion chamber (afterburning). In such ZhRD the gas, which passed through the turbine, heads on the gas conductor into the chamber/camera as one of the basic propellant components, the engines can work on the diagram "gas-liquid" and "gas-gas". Their general/common/total special feature/peculiarity is the high gas pressure at the turbine exhaust: it exceeds pressure p_k on the value of the hydraulic losses in the gas conductor and of pressure differential on the gas injectors of chamber/camera.

Besides the generator gas, produced in mono- or two-component ZhGG, as the working medium/propellant of turbine can serve the gas, which is generated as a result of heating one of the basic components of propellant (for example, hydrogen) in the coolant passage of chamber/camera.

ZhRD with one-component ZhGG it is possible to create, if one of the basic propellant components is capable to be decomposed/expanded with the liberation of heat.

Let us examine diagram of ZhRD, in which the working medium/propellant of turbine are the decomposition products of oxidizer (for example, peroxide of hydrogen H_2O_2) (Fig. 9.2). In the gas generator of this engine is supplied complete oxidizer consumption. The generating gaseous decomposition products enter turbine, and then on the gas conductor - into the combustion chamber.

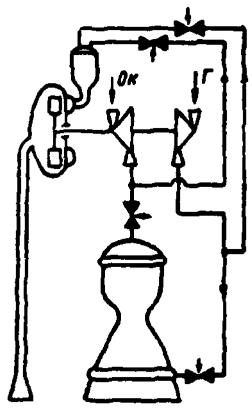

Fig. 9.1.

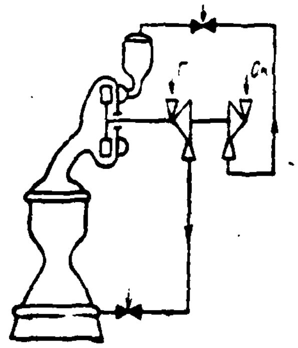

Fig. 9.2.

Fig. 9.1. ZhRD with throw-cut of exhaust generator gas into environment.

Fig. 9.2. ZhRD, which works on diagram "gas-liquid" with oxidative one-component ZhGG.

Page 133.

Fuel flows/occurs/lasts over the coolant passage of chamber/camera, cooling it, after which it in the liquid state is supplied inside the combustion chamber. The gas generator of this engine is called oxidative.

Is possible utilization of ZhRD with one-component reducing ZhGG, in this case into the combustion chamber are introduced liquid oxidizer and decomposition products of fuel (for example, ammonia NH_3 or hydrazine N_2H_4).

In ZhRD with oxidative two-component ZhGG (Fig. 9.3) into the latter is supplied complete oxidizer consumption from the pump and the relatively low part of the fuel; its basic part flows/occurs/lasts over the coolant passage and in the liquid state is introduced inside the chamber/camera, which in contrast to diagrams examined above is afterburner. Therefore such ZhRD call afterburners of generator gas.

To their number it relates also ZhRD with reducing two-component ZhGG (Fig. 9.4); into the chamber/camera of this engine enter the spent reducing generator gas and liquid oxidizer, and in ZhGG - complete fuel consumption (after the passage through the coolant passage of chamber/camera) and the relatively low part of the oxidizer.

Since pressure in ZhGG of the engines in question is more than pressure $p_{к}$ the pressure of that part of the propellant component, that heads in ZhGG, must be more than the pressure of basic part of the component, supplied directly to the chamber/camera. For this

purpose consecutively/serially after the basic pump (pump of first stage) is installed the additional ("pumping") pump, called also secondary pump (see Fig. 9.3 and 9.4).

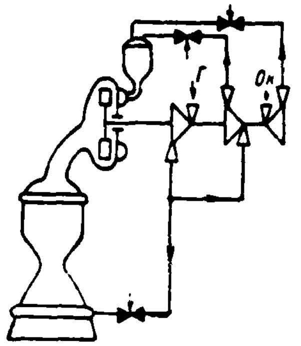

Fig. 9.3.

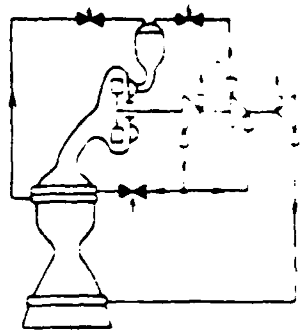

Fig. 9.4.

Fig. 9.3. ZhRD, which works on diagram "gas-liquid" with oxidative two-component ZhGG.

Fig. 9.4. ZhRD, which works on diagram "gas-liquid" with reducing two-component ZhGG.

Page 134.

Let us compare ZhRD with oxidative and reducing ZhGG. Usually for two-component ZhRD coefficient $\varkappa$ is more than one, i.e., oxidizer consumption is more than fuel consumption. The available power of turbine, as it will be shown in §13.13, depends on the gas flow through the turbine and on product RT of the gas indicated. Therefore from the point of view of the gas flow for the drive

turbines ZhRD with oxidative ZhGG have an advantage before ZhRD with reducing ZhGG. However, the oxidative gas, which has high temperature, exerts the strong oxidizing effect/action on the structural materials; therefore its temperature it is necessary to decrease. However, as a whole in ZhRD is more profitable to use oxidative ZhGG.

Product RT of reducing gas generator gas of hydrogen ZhRD due to the high gas constant of hydrogen has high value; therefore in them is more expedient to use reducing ZhGG.

ZhRD with the gasification of the working medium/propellant of turbine in the coolant passage of chamber/camera it is possible to create, if as fuels serves liquid hydrogen. In this case there is no need for in ZhGG, which simplifies the schematic of engine.

One of the possible schematics of this engine is shown in Fig. 9.5. Liquid hydrogen passes through two consecutively/serially established/installed pumps, after which it enters the coolant passage of chamber/camera. Formed gaseous hydrogen heads for the turbine, and then on the gas conductor - into the combustion chamber. Oxidizer (for example, liquid oxygen) is supplied into barrel; pump can be installed on the separate shaft and be brought with the aid of train of reducing gears from the shaft on which are

established/installed two hydrogen pumps and turbines.

The distinctive special feature/peculiarity of such ZhRD is the low turbine inlet gas temperature comprising approximately/exemplarily of 220-275°K. In the engines with ZhGG the temperature indicated is substantially above (≈800-1075°K).

Shortcoming of ZhRD with the gasification of the working medium/propellant of turbine in the coolant passage of chamber/camera is relatively low pressure $p_к$ (40-50 bars [≈40-50 kgf/cm²)].

In ZhRD, which work on the diagram "gas-gas" (Fig. 9.6), both of propellant components completely are used for the drive of turbopump units, while in ZhRD, which work on the diagram "gas-liquid", one of the components completely is not used for this purpose or is used its small part.

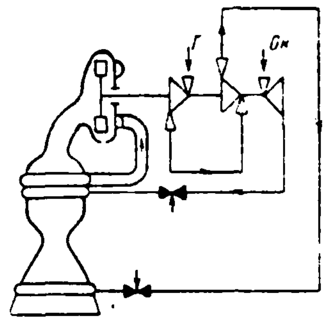

Fig. 9.5. ZhRD, which works on the diagram "gas-liquid" with the gasification of the working medium/propellant of turbine in the coolant passage of chamber/camera.

Page 135.

In ZhRD, which work on the diagram "gas-gas", there is on two TNA and ZhGG. The combustion products of reducing ZhGG 8 will become by the working medium/propellant of the turbine 7 of TNA of fuel; after turbine they are forwarded on gas conductor 6 for afterburner 1. Analogous with this the combustion products of oxidative ZhGG enter the turbine 3 of TNA of oxidizer, and then on gas conductor 2 - also into afterburner.

Pump 9 basic part of the fuel supplies into the reducing ZhGG, and smaller - into the oxidative. From pump 4 basic part of the oxidizer enters the the oxidative of ZhGG, and smaller - into the

reducing.

As is evident, ZhRD, which work on the diagram "gas-gas", are afterburners of generator gases in the chamber/camera. Such ZhRD can have the higher pressure of combustion products in afterburner in comparison with ZhRD, which work on the diagram "gas-liquid", or one and the same high pressure in afterburner at smaller necessary pressures of propellant components at the output/yield from the pumps.

ZhRD with the input/introduction of working medium/propellant after operation into the turbine into the expanding section of nozzle (Fig. 9.7). If engine works on the diagram "liquid-to-liquid", but working body after operation in the turbine is not thrown out into the environment, but it is introduced into the expanding section of nozzle, then specific jet firing increases/grows; however it is less than the specific impulse ZhRD, which work on the diagram "gas-liquid" or "gas-gas".

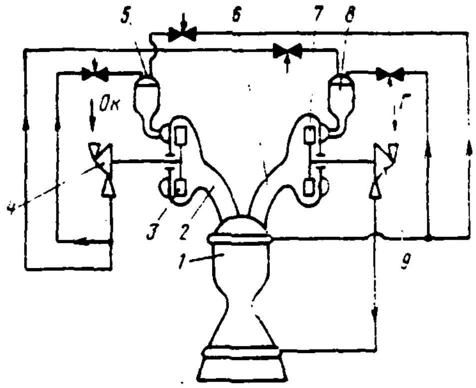

Fig. 9.6.

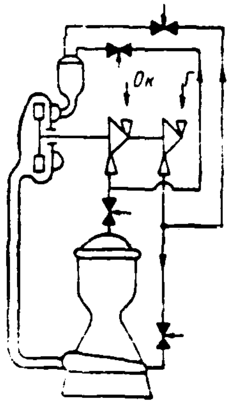

Fig. 9.7.

Fig. 9.6. ZhRD, working cn diagram "gas-gas": 1 - afterburner; 2, 6 - gas conductors; 3 - turbine of TNA cf oxidizer; 4 - pump of cxidizer; 5 - oxidative ZhGG; 7 - turbine of TNA of fuel; 8 - reducing ZhGG; 9 - fuel pump.

Fig. 9.7. ZhRD with input/introduction of working body after operation into turbine intc expanding section of nozzle.

Page 136.

The example of engine with the input/introduction of working medium/propellant after cperaticn intc the turbine into the expanding section of nozzle is ZhRD F-1 cf first stage cf American carrier

rocket "Saturn-5".

In engine installation with ZhRD enter the systems, the aggregates and the assemblies, which ensure:

a) arrangement/position and storage of liquid components of propellant (tanks);

b) their supply into chamber/camera;

c) engine starting;

d) propellant ignition (for the engines with the nonspontaneously combustible fuel/propellant);

e) cooling chamber/camera;

f) a change in engine power rating;

g) the creation of efforts/forces and moments/torques for the flight control of rocket vehicle;

h) engine cutoff.

Some systems in many respects are analogous for different thermal RD, while some - for all types of rocket engines. For example, for the creation of efforts/forces for the purpose of the flight control of rocket vehicle an engine of any type, including electrical, can be diverged to certain angle which causes the appropriate jet deflection.

The feed systems of liquid propellant components in the engine installations with ZhRD (see Chapter XIII), other thermal RD and are partly with the electric motors also analogous.

All types of rocket engines with the relatively high temperature of the products of combustion, decomposition, heating or plasma have a cooling system, i.e., branch system of the heat fluxes, which enter the chamber walls.

The mode of operation of the majority of rocket engines changes by changing the expenditure/consumption of working medium/propellant (for chemical RD - by changing the propellant component flow).

§9.2. Selection of optimal pressure

In §2.4 it was shown that for obtaining high characteristic velocity of rocket vehicle were necessary the high values of specific

impulse DU and ratio of the initial mass of rocket vehicle to the final.

The degree of the perfection of engine installation can be estimated by relation $I_\Sigma/m_{ДУ}$. Optimal is this pressure $p_к$, with which the relation indicated for assigned magnitude I_Σ it has great value.

Optimal pressure $p_к$ depends, first of all, on the feed system of propellant components into the chamber/camera.

For each type of pressururized-propellant feed (with the utilization of gas AD, ZhGG or TGG) with an increase in pressure $p_к$ to certain of its value relation $I_\Sigma/m_{ДУ}$ increases/grows, and with further growth $p_к$ it is decreased.

Page 137.

Let us explain the special feature/peculiarity indicated. We will proceed from conditions $m_{ДУ}$=const and p_c=const. For the thrust chamber is characteristic an increase in the specific impulse with an increase in pressure $p_к$, but in proportion to its increase an increase in the specific impulse significantly is retarded (see §5.4).

Simultaneously with an increase in pressure $p_к$ it is necessary to raise pressure in the tanks, which requires, in turn, increases in the wall thicknesses, and consequently, their mass. Furthermore, increases/grows the mass of the nozzle of chamber/camera in connection with an increase in values ε_c and $\bar{f}_c$. Therefore with an increase in pressure $p_к$ for the safeguard of condition $m_{ДУ}$ =const the mass of fuel/propellant in the tanks DU it is necessary to decrease.

An increase in relation $I_Σ/m_{ДУ}$ with an increase in pressure $p_к$ is explained by the fact that in this interval of pressure $p_к$ the specific impulse increases/grows intensively, and value $I_Σ$ is increased in spite of the decrease of the mass of fuel/propellant.

With an increase in pressure $p_к$ over the optimal relation $I_Σ/m_{ДУ}$ begins to be decreased, which indicates the greater effect of the decrease of the mass of fuel/propellant due to increase of mass of tanks and system of displacement in comparison with the effect of an increase in the specific impulse as a result of an increase in pressure $p_к$.

The less the mass of tanks and system of displacement for supplying the prescribed/assigned quantity of propellant components from the tanks into the engine chamber, the more the completion of

DU.

With the improvement of feed system increases/grows relation $I_{з}/m_{ду}$ and optimal pressure $p_{к}$. For example, pressure feed system with the utilization of ZhGG is more effective than system with gas ... (Fig. 3.8).

Usually pressure $p_{к}$ for ZhRD with the pressure feed system of propellant components is within the limits of 15-30 bars [≈15-30 kgf/cm²]. In ZhRD of space vehicles use in a number of cases lower pressures (7-8 bars [≈7-8 kgf/cm²)], which makes it possible to forego the external flowing cooling of chamber/camera and to attain the possibility of a significant power change, and also its high reliability.

For the engine installation with the pump feed system of propellant components into the chamber/camera also is an optimal pressure $p_{к}$, depending on a number of factors, including on the diagram DU.

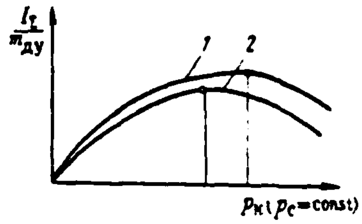

Fig. 9.8. Dependence of relation $I_\Sigma/m_{ду}$ on pressure $p_к$ for ZhRD with the pressururized-propellant feed with the gas storage tank of pressure (2) and ZhGG (1) ($m_{ду}$=const, p_c=const)

Page 138.

In the engine installation, which includes ZhRD with the throw-out of working medium/propellant after operation in the turbine into the environment, with an increase in pressure $p_к$ is increased the necessary pressure of propellant components at the output/yield from the pumps, which causes the need for an increase in the power of turbine (in §13.13 it will be shown that to increase the power of turbine is possible in essence by an increase in the gas flow through it; however in this case it descends specific jet firing, see §9.1).

In certain range of pressure $p_к$ in proportion to its growth specific jet firing increases/grows: an increase in the specific impulse of chamber/camera due to an increase in pressure $p_к$ exceeds

the decrease of specific jet firing as a result of an increase in the gas flow through the turbine.

At certain pressure $p_к$ is provided the greatest specific jet firing, while with further increase $p_к$ it begins to be decreased (Fig. 9.9). In this case a reduction/descent in the specific jet firing due to an increase in the gas flow through the turbine exceeds growth in the specific impulse as a result of an increase in pressure $p_к$.

The selection of optimal pressure $p_к$ and for ZhRD with the pump feed system must be produced not of the condition of the greatest specific jet firing, but from the condition for greatest relation $I_у/m_{ду}$.

In the engine installation with ZhRD, which work on the diagram "gas-liquid" or "gas-gas", the specific impulse of chamber/camera and engine is one and the same value. For such engines with an increase in pressure $p_к$ simultaneously with an increase in the specific impulse grows the mass of chamber/camera.

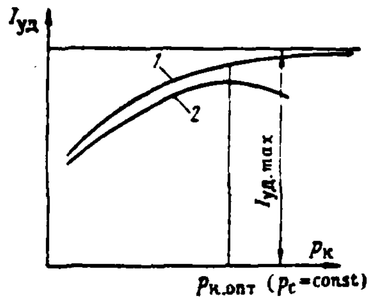

Fig. 9.9.

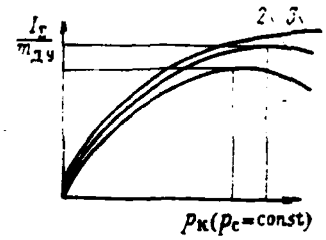

Fig. 9.10.

Fig. 9.9. Dependence of specific impulse of chamber/camera (1) and engine (2) with throw-out of working medium/propellant after operation in turbine into environment on pressure $p_к$ (p_c=const)

Fig. 9.10. Dependence of relation $I_г/m_{ду}$ for ZhRD with throw-out of generator gas into environment with ZhGG on auxiliary components of propellant (1), on basic components of propellant (2) and for ZhRD with afterburning of generator gas (3).

Page 139.

Therefore also there is an optimal pressure $p_к$, corresponding to greatest relation $I_г/m_{ду}$.

The more modern the feed system of propellant components into

the chamber/camera and the diagram cf ZhRD, the greater the relation $I_{y}/m_{\text{ДУ}}$, moreover it is increased with an increase in pressure $p_{к}$ (Fig. 9.10). Best are DU with ZhRD, which work on scheme "gas-liquid" or "gas-gas". It is especially profitable to use such engines at high pressures $p_{к}$ for DU cf high thrust.

In §5.4 it was shown that with increase $p_{к}$ are decreased the sizes/dimensions cf chamber/camera and is simplified its manufacture.

High-pressure use/application $p_{к}$ is connected with some difficulties during the creation cf engine. They include: the need for more effective cooling, difficulties of the safeguard of airtightness cf joints, and alsc strength and efficiency of engine accessories. However, these difficulties successfully are overcome. Shortcomings in high-pressure utilization $p_{к}$ are also an increase in the ccst/value of engines and certain reduction/descent in their reliability.

Pressure $p_{к}$ for the majority cf contemporary ZhRD with the pump feed system equally to 50-100 bars [≈50-100 kgf/cm²]; for some ZhRD - to 200 bars [≈200 kgf/cm²]; is investigated worthwhileness of higher values $p_{к}$ of 280-350 bars [≈280-350 kgf/cm²] and more.

Chapter X.

LIQUID CHEMICAL PROPELLANTS.

In §1.2 it was indicated that chemical fuel/propellant are called the substances which with the entrance into the chemical reaction isolate heat and form in essence gaseous products. Most typical chemical fuels/propellants consist of oxidizer and fuel. Oxidizer is called the substance, which consists mainly of the oxidative elements/cells, and by fuel - from the combustible elements/cells. In the process of chemical reaction occurs the electron transfer in the outermost electronic shell of the atoms: the atoms of combustible elements/cells give up their electrons to the atoms of oxidative elements/cells.

The component of chemical fuel/propellant (Fig. 10.1) is called the liquid substance, stored in the separate tank and supplied on the separate main into the engine chamber. The component of chemical fuel/propellant can be the solid, placed directly in the chamber/camera. As the component of chemical fuel/propellant can serve also the mixture either of liquid or solid individual substances, and also the mixture of liquid and solid individual substances.

Page 140.

In certain cases into the propellant component introduce the special additives (from the portions of percentage to several percentages) for the purpose of an improvement in its any property.

The liquid propellant components, which contain solid metallic particles, call metal-containing, or metallized; they distinguish two types of these components: suspension and colloidal solutions. Suspension is called the liquid component in volume of which evenly distributed the fine/small solid particles of metal. Colloidal solution differs from suspension in terms of the smaller size of the particles of the metal.

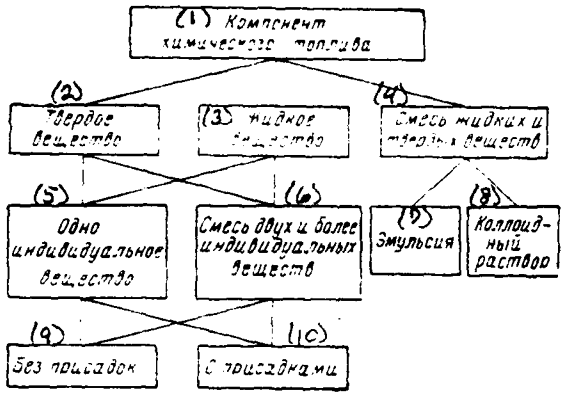

Fig. 10.1. Classification of the components of chemical fuel/propellant.

Key: (1). Component of chemical fuel/propellant. (2). Solid. (3). Liquid substance. (4). Mixture of liquid and solids. (5). One individual substance. (6). Mixture of two or more individual substances. (7). Emulsion. (8). Colloidal solution. (9). Without additives. (10). With additives.

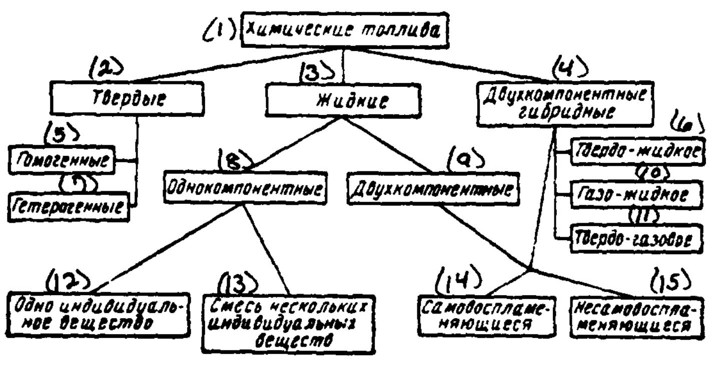

Fig. 10.2.

Fig. 10.2. Classification of chemical fuels/propellants.

Key: (1). Chemical fuels/propellants. (2). Solid. (3). Liquid. (4). Two-component hybrid. (5). Homogeneous. (6). Solid-liquid. (7). Heterogenic. (8). One-component. (9). Two-component. (10). gas-liquid. (11). solid-gas. (12). One individual substance. (13). Mixture of several individual substances. (14). Igniting spontaneously. (15). Combusting nonspontaneously.

Page 141.

Chemical fuels/propellants (Fig. 10.2) classify according to the following signs:

a) in a quantity of basic components - (mono-, two- and three-component fuels/propellants);

b) due to the state of aggregation of basic components of propellant - (solid, liquid and solid-liquid (hybrid) propellants);

c) on the special features/peculiarities of the reaction of components of propellant with their direct contact (igniting spontaneously and nonspontaneously combustible fuels/propellants).

The hypergolic fuels will be ignited through $(3-8)\cdot 10^{-3}$ s after the contact of their components (the time indicated is called the period of the delay of spontaneous combustion). For the ignition of the nonspontaneously combustible fuels/propellants is required special system.

A number of three-component fuels/propellants includes, in particular, the fuels/propellants in which are included the oxidizer, combustible and component with the low value μ (for example, liquid hydrogen), which occasionally referred to as diluent.

Solid fuels can be homogeneous (uniform) and heterogenic (heterogeneous or mixture). Homogeneous fuel/propellant is the chemical substance whose molecule contains oxidizer, and combustible; homogeneous fuel/propellant is also the solid solution of two such chemical substances. Composite propellant - mechanical mixture of oxidizer (usually crystal) and fuel which simultaneously plays the role of binder, providing thereby the creation of solid-propellant grain with the necessary mechanical characteristics. Solid fuels are examined in chapter XVI.

§10.1. Simple oxidizers and fuels.

The components of chemical fuels/propellants contain both

oxidative and combustible elements/cells. The propellant components, which consist of one type oxidative or combustible elements/cells, call respectively simple oxidizers or fuels.

The oxidizers include oxygen and halogens: fluorine, chlorine, bromine and iodine. The greatest oxidizing ability they possess oxygen and especially fluorine. Them are used in the form of simple oxidizers and in connection with other less effective oxidative elements/cells. Some properties of simple oxidizers are given in Table 10.1.

The basic fuels of chemical rocket propellants are hydrogen, lithium, beryllium, boron, carbon, magnesium, aluminum and silicon; other combustible elements/cells - sodium, calcium, phosphorus, titanium, zirconium - are less effective. In Table 10.2 are given the basic properties of simple fuels.

The fuels/propellants in which as the oxidizer is used fluorine, are more effective than fuels/propellants on the basis of oxygen.

Page 142.

This is explained by the following special features/peculiarities of oxides (the end products of the reaction of fuels with oxygen) and

fluorides (the end products of the reaction of fuels with fluorine).

1. Heat of formation of fluorides for majority of combustible elements/cells examined is more than heat of formation of oxides.

2. Boiling point and melting of fluorides is substantially lower than appropriate temperature of oxides. Therefore in the majority of the cases fluorides desert the nozzle of chamber/camera chemical RD in the gaseous state, and many oxides (especially BeO and Al_2O_3) in the liquid or solid state.

Hydrogen during the reaction with oxygen and fluorine gives not the highest heat of formation of corresponding oxide (H_2O) and fluoride (HF), but these compounds possess low molecular weight and low values $T_{кип}$ and $T_{пл}$, which makes the fuels/propellants in which is used oxygen and fluorine as the oxidizer and hydrogen as the fuel, by very effective ones.

Highly efficient fuels are also metals and metalloids with the low molecular weight (Li, Be, B). Carbon relates to a number of relatively barely effective combustible elements/cells.

Table 10.1. Some properties of simple oxidizers [35], [37].

(1) Окислитель	(2) Химическая формула	(3) Порядковый номер элемента	μ, (6) $\frac{кг}{кмоль}$	(4) Плотность в жидком состоянии, (7) $\frac{кг}{м^3}$	$T_{пл}$	$T_{кип}$
					(5) при нормальном давлении, °K	
(8) Кислород	O_2	8	31,999	1144 (9) (при $T_{кип}$)	54,35	90,18
(10) Фтор	F_2	9	37,997	1507 (9) (при $T_{кип}$)	53,53	85,02
(11) Хлор	Cl_2	17	70,906	1557 (9) (при $T_{кип}$)	171,85	238,45; 239,05
(12) Бром	Br_2	35	159,808	3102 (при (9) 298°, 15K)	265,85	331,05; 331,93
(13) Иод	J_2	53	253,809	3960 (при (9) 393°, 15K)	386,85	455,95; 457,50

Key: (1). Oxidizer. (2). Chemical formula. (3). Reference number of element/cell. (4). Density in the liquid state. (5). at normal pressure. (6). kg/kmole. (7). kg/m³. (8). Oxygen. (9). with. (10). Fluorine. (11). Chlorine. (12). Bromine. (13). Iodine.

Page 143.

Table 10.2. Some properties of simple fuels [35], [37].

(1) Горючее	(2) Химическая формула	(3) Порядковый номер	μ	ρ (4) в твердом состоянии	ρ (5) в жидком состоянии	$T_{пл}$	$T_{кип}$
						(6) при нормальном давлении	
			(7) кг/кмоль	(8) кг/м³		°K	
(9) Водород	H_2	1	2,016	—	(10) 70,97 (при $T_{кип}$)	13,94	20,39
(11) Литий	Li	3	6,939	534*	507 (10) (при 473°, 15K)	453,65	1620,15
(12) Бериллий	Be	4	9,012	1850*	—	1556,15	2757,15
(13) Бор	B	5	10,811	2300*	—	2300,15	3950,15
(14) Углерод (графит)	C	6	12,011	2250	—	3873; 3973	≈4473
(15) Магний	Mg	12	24,305	1740*	—	923,15	1381,15
(16) Алюминий	Al	13	26,982	2700*	2289 (10) (при 1273°, 15K)	932,15	2740,15
(17) Кремний	Si	14	28,086	2000 (18) (аморфный)	—	—	≈2873

Key: (1). Combustible. (2). Chemical formula. (3). Reference number. (4). in solid state. (5). in the liquid state. (6). at normal pressure. (7). kg/kmole. (8). kg/m³. (9). Hydrogen. (10). with. (11). Lithium. (12). Beryllium. (13). Boron. (14). Carbon (graphite). (15).

Magnesium. (16). Aluminum. (17). Silicon.

FOOTNOTE [1]. At a temperature of 298°, 15 K. ENDFOOTNOTE.

§10.2. Special requirements for the liquid chemical propellants.

General and specific requirements for the chemical fuels/propellants are examined in §3.2 and 3.3.

In accordance with equation (5.10) of fuel/propellant must provide the high values of thrust coefficients K_P (see §5.3) and β (see §4.5). The coefficients indicated as rate W_c (see §4.5), increase/grow with an increase in temperature and gas constant of products combustion at the nozzle entry, and also with an increase in the expansion ratio ε_c and the decrease of index n_p.

Page 144.

Essential is the effect of temperature $T_к$, which depends on net calorific power $H_{раб}$, of that determined by type and relationship/ratio of propellant components.

The components of chemical fuels/propellants analogous with all working medium/propellants RD (see §3.2) must possess high density. Especially this is important for the oxidizer, since its density in essence determines fuel density.

In connection with the fact that on the characteristic velocity of rocket vehicle values $I_{уд}$ and $\rho_т$ have different effect, appears need in the combined estimated parameter. Such parameter is expression $I_{уд}\rho_т^c$ where c - index whose value is determined according to the equation

$$c=\frac{m_т/m_{нач}}{\lg\frac{1}{1-m_т/m_{нач}}}.$$

Here $m_т$ - mass of propellant components.

To the maximum of the characteristic velocity of rocket vehicle corresponds the great value of expression $I_{уд}\rho^c_т$ Index c, which is determining the density effect of fuel/propellant on the characteristic velocity of rocket vehicle, is lower than unity. Therefore specific impulse influences characteristic velocity more than fuel density. With the decrease of index c (i.e. with an increase in relation $m_т/m_{нач}$) the density effect $\rho_т$ is decreased. At value $m_т/m_{нач}=0{,}8$, characteristic for the ballistic missiles, c=0.5.

For the upper stages of rockets the density effect of

fuel/propellant is decreased, and the effect of specific impulse increases/grows. Therefore for the upper stages of rockets is recommended the use/application of a fuel/propellant liquid oxygen + liquid hydrogen, in spite of the extremely low density of liquid hydrogen ($\rho \approx 71$ kg/m³).

The important characteristics of chemical fuel/propellant are also the stability (stability) of the course of the reaction of burning or decomposition and starting/launching properties.

The stability of burning or decomposition of fuel/propellant is determined in essence by the amplitude of fluctuations of pressure $p_к$: is more, the less stably occur chemical reactions in the chamber/camera and the lower the reliability of its operation (see §15.1).

Fuels/propellants with good starting/launching properties provide stable (without the large oscillations pressures $p_к$) the start-up conditions of engine. For example, bipropellants with good starting/launching properties easily and smoothly will be ignited in broad limits of a change of coefficient $\varkappa$, which is explained by their following special features/peculiarities:

a) by light volatility;

b) by low ignition temperature.

Page 145.

c) by the small heat, necessary for the inflammation;

d) by the short ignition delay $\tau_{з.в}$.

From the point of view of the safeguard of good starting/launching properties and stability of the process of burning, and also simplification in the construction/design of engine the hypergolic fuels usually have advantages before those combusting nonspontaneously.

To the liquid propellant components present the following additional requirements:

a) low viscosity/ductility/toughness, also, as far as possible its small change in the temperature range of the operation of engine;

b) small surface tension;

c) the low saturated vapor pressure.

With the low viscosity/ductility/toughness of propellant components is decreased the hydraulic resistance of the mains of engine, which leads to the decrease of the necessary expenditures of power for the supply of components into the chamber/camera.

With the low viscosity/ductility/toughness and the small surface tension of liquid propellant components is improved their atomization, i.e., the fragmentation to the smallest particles during the input/introduction of components into the chamber/camera, which facilitates more complete combustion.

The low saturated vapor pressure of propellant components decreases the losses to their vaporization and is exerted favorable influence on some other parameters of engine and rocket vehicle as a whole.

If engine chamber has external flowing cooling, then one of the propellant components must possess good cooling properties.

It is necessary to indicate that there are no such fuels/propellants which it would be possible to use with the identical effect for the rocket engines of different

designation/purpose and with different thrust. Therefore the selection of fuel/propellant in each specific case must be produced very thoroughly.

§10.3. Characteristic of liquid propellants.

In ZhRD in essence are used the bipropellants. Such fuels/propellants call also bipropellant fuels, since oxidizer and combustible are stored in the separate tanks and are supplied into the chamber/camera on the different mains.

Are more simple by the construction/design and in the operation of ZhRD, which work on the one-component (or unitary) fuel/propellant.

The monopropellants, which are the mixture of oxidizer and fuel or the solutions of fuel in the oxidizer, can possess sufficiently high energy characteristics, but such fuels/propellants are inclined to the explosion. The same can be said about the monopropellant, which consists of one substance into molecule of which they enter both oxidative and combustible elements/cells (for example, nitromethane CH_3NO_2).

Page 146.

The monopropellants, which consist of one individual substance (for example, hydrazine N_2H_4) and which isolate heat as a result of decomposition of the presence of catalyst, are sufficiently stable, but have comparatively low energy characteristics.

Are used both solid and liquid catalysts. Solid catalyst is placed directly into the chamber/camera; its mass with the work of engine virtually does not charge.

Liquid catalyst is placed in the separate tank and continuously is supplied into the chamber/camera on the special conduit/manifold.

The example of starting/launching propellant component for ZhRD is triethylaluminum $Al(C_2H_5)_3$ - the liquid, which is ignited in air; it are used for the ignition of the nonspontaneously combustible fuels/propellants.

During the design of ZhRD and any other type of RD they strive as far as possible to exclude starting/launching and additional propellant components. The use/application only of basic propellant components simplifies the construction/design of rocket apparatus and servicing devices of starting buildings, facilitates servicing tanks,

etc.

It was above indicated that into the rocket propellant in a number of cases are introduced the additives. Are used, in particular, the additives, which ensure:

a) the prolonged chemical stability of component of propellant (inhibitors);

b) a reduction/descent in the corrosiveness of component of propellant (deactivators);

c) the decrease of value $\tau_{з.в}$ (catalysts);

d) the spontaneous combustion of the fuel/propellant (this it is possible to attain, for example, by the introduction of liquid fluorine into liquid oxygen).

§10.4. Liquid oxidizers and the fuels of rocket propellants.

Oxidizers usually compose the large part of the fuel/propellant throughout the mass. Wholly of oxidative elements/cells consist simple oxidizers (O_2, F_2) or compounds from the oxidative elements/cells (oxide of fluorine OF_2, fluorides of halogens: ClF_3,

ClF_5, BrF_3, BrF_5, JF_5, etc., perchloryl fluoride ClO_3F, etc.).

Some oxidizers contain in the molecule together with the oxidative element/cell nitrogen, which is the neutral element/cell (nitrogen tetroxide N_2O_4, fluorides of nitrogen NF_3 and N_2F_4, etc.) or combustible and neutral elements/cells simultaneously (nitric acid HNO_3, tetranitromethane $C(NO_2)_4$, perchloric acid $HClO_4$, etc.).

In peroxide of hydrogen H_2O_2 are included oxidative and combustible elements/cells.

Page 147.

Most widely in ZhRD are used the following oxidizers: oxygen, nitric acid, nitrogen tetroxide and peroxide of hydrogen. The physicochemical properties of basic oxidizers are given in Table 10.3.

Are most effective the fuels, which are whole of the combustible elements/cells. The presence of nitrogen or oxidative element/cell in the composition of fuel, as a rule, decreases energy propellant properties.

The utilized and promising fuels of liquid propellants can be

subdivided into the following groups:

1. Liquid hydrogen and the hydrazoic fuels: hydrazine N_2H_4, ammonia NH_3.

2. Fuels, which contain in their composition hydrogen, nitrogen, carbon and being derivatives of hydrazine: monomethyl hydrazine (MMG) $H_2N-NH(CH_3)$, unsymmetrical dimethylhydrazine (UDMH) $H_2N-N(CH_3)_2$ and aerozine-50, which is mixture (1:1 throughout mass) of hydrazine and UDMH.

3. Hydrocarbon fuels: kerosene (mixture of hydrocarbons, obtained with distillation of petroleum); methane CH_4 (liquified hydrocarbon, which is fundamental component of natural gas); ethane C_2H_6 propane C_3H_8 (also liquified hydrocarbons), etc.

4. Fuels, which contain in their composition hydrogen, carbon and oxygen (alcohols): ethyl alcohol C_2H_5OH, methyl alcohol CH_3OH, etc.

The physicochemical properties of basic fuels are given in Table 10.4.

§10.5. Characteristics of bipropellants.

The characteristics of basic bipropellants are given in Table 10.5.

In Table 10.6 are noted the igniting spontaneously and nonspontaneously combustible fuels/propellants.

Fuels/propellants on the basis of liquid oxygen. In the initial period the developments of ZhRD and during the years of the Second World War used extensively a fuel/propellant liquid oxygen O_2+ethyl alcohol C_2H_5OH. A comparatively low heating power of this fuel/propellant led to the replacement by its fuel/propellant O_2+kerosene.

Fuel/propellant O_2+kerosene is cheap and reliable, it is well mastered in the production and the operation. Some difficulties in cooling of chamber/camera due to the high temperature of the products of combustion and comparatively small quantity of fuel in the propellant composition successfully are overcome, and fuel/propellant O_2+kerosene extensively is used in contemporary ZhRD. Are developed ZhRD, which develop on this fuel/propellant thrust to 7000 kn [≈700 T].

Pages 148-149.

Table 10.3.

(1) Окислитель	(2) Химическая формула	μ	(3) $T_{пл}$ при нормальном давлении	$T_{кип}$	ρ *	i_0 **	(4) Химическая стабильность	(5) Предельно допустимая концентрация в воздухе
		(6) $\frac{кг}{кмоль}$	°K		(7) $\frac{кг}{м^3}$	(8) $\frac{кдж}{кг}$		(9) $\frac{мг}{м^3}$
(10) Кислород	O_2	31,999	54,35	90,18	1144	—398	(11) Стабилен	(12) Нетоксичен
(13) Перекись водорода	H_2O_2	34,015	272,26	423,35	1442	—5530	(14) Нестабилен	1,0
(15) Азотная кислота	HNO_3	63,014	231,56	357,25	1504	—2753		5,0
(16) Четырехокись азота	N_2O_4	92,011	261,95	294,3	1442	—209	(17) Стабильна	5,0
(18) Фтор	F_2	37,997	53,53	85,02	1507	—335	(11) Стабилен	0,03
(19) Окись фтора	OF_2	53,996	49,35	127,85	1521	222	(17) Стабильна	0,01
(20) Трифторид хлора	ClF_3	92,448	196,83	284,90	1809	—2000	(11) Стабилен	(21) Очень токсичен

(22) Пентафторид хлора	ClF_5	130,457	—	—	1750	—	•	(23) Слабо токсичен
(24) Трифторид азота	NF_3	71,008	66,36	144,14	1531	—2050	•	(21) Очень токсичен
(25) Тетрафторгидразин	N_2F_4	104,016	105,15	200,15	(26) 1500 (при 173° K)	—	•	(27) То же
(28) Трифторид брома	BrF_3	136,916	281,92	398,9	2797	—	•	(29) Токсичен
(30) Пентафторид брома	BrF_5	174,916	210,65	313,45	2465	—2625	•	•
(31) Перхлорилфторид	ClO_3F	102,457	125,41	226,48	1691	—398	•	(23) Слабо токсичен
(32) Хлорная кислота	$HClO_4$	100,465	161,15	403,15	1772	—460	(33) Нестабильна	(29) Токсична

Key: (1). Oxidizer. (2). Chemical formula. (3). at normal pressure. (4). Chemical stability. (5). Maximum permissible concentration in air. (6). kg/kmole. (7). kg/m³. (8). kJ/kg. (9). mg/m³. (10). Oxygen. (11). It is stable. (12). It is nontoxic. (13). Peroxide of hydrogen. (14). It is unstable. (15). Nitric acid. (16). Nitrogen tetroxide. (17). It is stable. (18). Fluorine. (19). Fluorine oxide. (20). Chlorine trifluoride. (21). It is very toxic. (22). Pentafluoride of chlorine. (23). It is weakly toxic. (24). Trifluoride of nitrogen. (25). Tetrafluorinehydrazine. (26). with. (27). the same. (28). Trifluoride of bromine. (29). It is toxic. (30). Bromine pentafluoride. (31). Perchloryl fluoride. (32). Perchloric acid.

(33). It is unstable.

FOOTNOTE [1]. Values at a temperature of 298°, 15K, and for the low-boiling oxidizers - at a boiling point.

[2]. Values at temperature of 293°K, and for low-boiling oxidizers - at boiling point. ENDFOOTNOTE.

Page 150.

By the best control characteristics even more steady-state burning in comparison with the fuel/propellant O_2+kerosene possess fuels/propellants O_2+NH_4, O_2+NH_3, O_2+MMG and O_2+UDMH. From them most extensively is used the fuel/propellant O_2+UDMH. For these fuels/propellants, and also for fuel/propellant O_2+H_2 in spite of their high heating power is characteristic a reduced temperature of combustion products, which facilitates cooling chamber/camera.

Table 10.4. Some properties of liquid fuels [2], [23], [35].

(1) Горючее	(2) Химическая формула	μ	$T_{пл}$ (3) при нормальном давлении	$T_{кип}$ (3) при нормальном давлении	ρ *	$i_п$ **	(4) Химическая стабильность	(5) Предельно допустимая концентрация в воздухе
		(6) кг/кмоль	°К	°К	(7) кг/м³	(8) кдж/кг		(9) мг/м³
(10) Водород	H_2	2,016	13,94	20,39	70,97	—3828	(11) Стабилен	(12) Нетоксичен
(13) Гидразин	N_2H_4	32,048	274,68	386,65	1004	1573	.	(14) Токсичен
(15) Аммиак	NH_3	17,032	195,39	239,73	682	—4180	.	20—50
(16) Монометилгидразин	$H_2N—NH(CH_3)$	46,075	220,75	360,65	874	1222	.	(14) Токсичен
НДМГ	$H_2N—N(CH_3)_2$	60,102	215.95	336.25	784	774	.	.
(17) Аэрозин-50	—	45,584	265.85	343,25	899	1173	.	.
Керосин (18)	$C_{7,21}H_{13,29}$ (19) (условная формула)	—	200—220	450	820—850	—1728	.	300
(20) Метан	CH_4	16,047	89,15	111,65	424	—5439	.	(21) Слабо токсичен
(22) Этиловый спирт	C_2H_5OH	46,070	159,05	351,47	785	—6025	.	1000
(23) Диборан	B_2H_6	27,67	107.65	180,65	430	438	(24) Стабилен в герметичном баке	(25) Весьма токсичен
(26) Пентаборан	B_5H_9	63,27	226,34	335,15	618	381	(27) То же	0,01

Key: (1). Combustible. (2). Chemical formula. (3). at normal pressure. (4). Chemical stability. (5). Maximum permissible concentration in air. (6). kg/kmole. (7). kg/m³. (8). kJ/kg. (9). mg/m³. (10). Hydrogen. (11). It is stable. (12). It is nontoxic. (13). Hydrazine. (14). Toxic. (15). Ammonia. (16). Monomethyl hydrazine. (17). Aerozine-50. (18). Kerosene. (19). conventional formula. (20). Methane. (21). It is weakly toxic. (22). Ethyl alcohol. (23). Diborane. (24). It is stable in pressurant storage

tank. (25). It is very toxic. (26). Pentaborane. (27). The same.

FOOTNOTE [1]. Values at a temperature of 298°, 15K, and for the low-boiling fuels - at a boiling point.

[2]. Values at temperature of 293°K, and for low-boiling fuels - at boiling point. ENDFOOTNOTE.

Page 151.

The greatest specific impulse (to 4800 N•s/kg [≈480 kg•s/kg]) of all contemporary familiar fuels/propellants provides fuel/propellant liquid oxygen + liquid hydrogen (O_2+H_2). Are developed ZhRD, which develop on the fuel/propellant indicated thrust to 1000 kN [≈100 T]; in the USA is conducted the work on the creation ZhRD with the thrust to 7000 kn [≈700 T]. In spite of the low density of fuel/propellant O_2+H_2 ($\rho_T \approx 320$ kg/m³) its use/application for ZhRD of upper booster stages makes it possible to significantly increase the mass of payload.

With the addition of liquid fluorine into liquid oxygen in a quantity to 5o/o all fuels/propellants on the basis of liquid oxygen

become igniting spontaneously.

With refuelling O_2+H_2 by fuel/propellant (70c/c O_2+30c/c F_2)+H_2 specific jet firing is increased. Mixture O_2+F_2 can be used with UDMH, kerosene and liquified hydrocarbons (methane, ethane and propane).

Fuels/propellants on the basis of peroxide of hydrogen. Peroxide of hydrogen was used extensively as the oxidizer in ZhRD during the Second World War.

However, in that period peroxide of hydrogen was used in the form of 80c/c aqueous solution, which decreased the heating power of fuel/propellant. In proportion to the development of the methods of stabilization of peroxide of hydrogen appeared the possibility to increase its concentration to 90c/c, and in certain cases to 98c/o.

Fuels/propellants on the basis of highly concentrated peroxide of hydrogen are not inferior to fuels/propellants on the basis of nitric acid on the density and at the same time provide somewhat larger specific impulse at a substantially smaller combustion temperature. By the additional advantage before the nitric acid and oxidizers on its basis is the smaller corrosiveness of peroxide of hydrogen.

Most is used extensively fuel/propellant H_2O_2+kerosene; more rarely they use: H_2O_2+UDMH, $H_2C_2+NH_3$ and $H_2O_2+N_2H_4$. Concentration H_2O_2 in all fuels/propellants indicated is equal to 90o/o. To a number of advanced propellants on the basis of peroxide of hydrogen relate $H_2O_2+B_2H_6$ and especially $H_2C_2+B_5H_9$. The important advantage of latter/last fuel/propellant is the fact that it consists of the high-boiling components.

Fuels/propellants on the basis of nitric acid. The heating power of such fuels/propellants is less than the heating power of fuels/propellants on the basis of liquid oxygen, but in contrast to the latter they possess high density and can long time be stored in the completely charged/filled rocket.

Nitric acid of 100o/o concentration is unstable product.

Page 152.

Table 10.5. Theoretical characteristics of some bipropellants [35] ($p_к$=68,946 of bar [68.046 phys. atm.]; p_c=1,013 bar [1 phys. at.]; $\varkappa=\varkappa_{опт}$; $p_h=p_3$; equilibrium expansion).

(1) Характе-ристика	(2) Горючее	(3) Окислитель									
		O_2	H_2O_2	HNO_3	N_2O_4	F_2	OF_2	ClF_5	ClF_3	N_2F_4	ClO_3F
$\varkappa$	H_2	4,00	7,33	6,14	5,25	8,09	5,67	11,50	11,50	11,50	6,14
	N_2H_4	0,92	2,03	1,50	1,33	2,37	1,50	2,71	2,70	3,25	1,50
	НДМГ	1,70	4,26	3,00	2,57	2,45	2,69	2,85	3,00	3,17	2,70
	B_5H_9	2,12	2,23	3,00	3,00	4,56	4,00	6,69	7,33	6,69	3,76
$\rho_т$ $\frac{кг}{м^3}$ (4)	H_2	284	435	393	353	468	375	605	616	517	403
	N_2H_4	1065	1261	1254	1217	1314	1263	1458	1507	1105	1327
	НДМГ	976	1244	1223	1170	1190	1214	1325	1381	1028	1288
	B_5H_9	897	1021	1107	1084	1199	1179	1413	1493	1027	1239
$T_к$ °K	H_2	2977	2419	2474	2640	3988	3547	3705	3434	3814	3003
	N_2H_4	3406	2927	3021	3247	4727	4047	4157	3901	4481	3467
	НДМГ	3608	3008	3147	3415	4464	4493	4003	3799	4226	3657
	B_5H_9	4160	2969	3588	3913	5080	5009	4656	4447	4840	4242
n_p	H_2	1,232	1,247	1,258	1,260	1,237	1,239	1,260	1,061	1,259	1,255
	N_2H_4	1,164	1,187	1,199	1,191	1,177	1,166	1,220	1,237	1,061	1,180
	НДМГ	1,144	1,160	1,172	1,166	1,149	1,167	1,189	1,198	1,186	1,171
	B_5H_9	1,111	1,112	1,128	1,121	1,150	1,136	1,150	1,152	1,155	1,125

β (5) $\frac{н \cdot сек}{кг}$	H_2	2431,1	2011,3	2001,5	2134,9	2553,7	2562,5	2145,7	2022,1	2272,2	2153,5
	N_2H_4	1890,7	1753,4	1714,2	1779,9	2212,4	2091,8	1928,0	1827,0	2056,4	1794,6
	НДМГ	1856,4	1714,2	1654,4	1725,0	2087,8	2136,0	1826,0	1726,0	1959,4	1754,4
	B_5H_9	1894,6	1832,9	1753,4	1781,9	2175,1	2165,3	1859,3	1751,5	2013,3	1784,8
$I_{уд.з}$ $\frac{н \cdot сек}{кг}$ (5)	H_2	3835,4	3161,7	3135,2	3341,1	4038,4	4042,3	3363,7	3145,0	3564,7	3373,5
	N_2H_4	3068,5	2813,5	2737,0	2854,7	3573,5	3392,1	3059,7	2889,0	3282,3	2895,9
	НДМГ	3037,1	2782,1	2671,3	2796,9	3411,7	3453,9	2925,3	2759,6	3147,9	2840,0
	B_5H_9	3135,2	3031,2	2880,2	2935,1	3539,2	3546,1	3026,3	2846,9	3275,4	2935,1
$I_{уд.п}$ $\frac{н \cdot сек}{кг}$ (5)	H_2	4471,8	3675,5	3636,3	3872,6	4683,7	4690,5	3887,4	3620,6	4118,8	3909,9
	N_2H_4	3625,5	3310,7	3210,7	3353,8	4210,0	4009,9	3567,7	3352,9	3827,5	3406,8
	НДМГ	3610,8	3294,1	3153,8	3304,8	4050,1	4085,5	3441,2	3239,1	3696,1	3351,9
	B_5H_9	3777,5	3659,8	3454,9	3526,5	4224,7	4248,2	3616,7	3402,9	3899,1	3525,5

Key: (1). Characteristic. (2). Ccmbustible. (3). Oxidizer. (4). kg/m³. (5). N•s/kg.

Page 153.

Therefore in ZhRD use the concentrated nitric acid, which contains about 2c/o H_2O and 0.5o/c of cxides of nitrogen NO_2 (this acid is called white fuming), or the solution of concentrated nitric acid and nitrogen tetroxide N_2C_4 (this sclution is called the red fuming acid). Latter/last oxidizer is more effective. The fuel/propellant on its basis, which uses UDMH as the fuel, is the example of the igniting spontaneously storable propellants with good control characteristics and steady-state burning.

However, fuels/propellants on the basis of nitric acid in essence are displaced by fuels/propellants on the basis of nitrogen tetroxide.

Fuels/propellants on the basis of nitrogen tetroxide. Most use extensively, including if necessary for prolonged storage, fuel/propellant N_2O_4+N_2H_4, N_2C_4+MMG and especially N_2O_4+aercsine-50 and N_2O_4-UDMH. They are somewhat inferior to fuel/propellant O_2+kerosene on the developed with engine specific impulse, but they exceed it on the density.

Fuels/propellants N_2O_4+aerosine-50 and N_2O_4+UDMH make it possible to create reliably working ZhRD with the high specific impulse and with the very high thrust in one chamber/camera.

Page 154.

For the engines with the relatively small thrust are used fuels/propellants $N_2O_4+N_2H_4$ and N_2O_2+MMG; the best starting/launching properties possesses the latter of fuel/propellant.

Fuel/propellant on the basis of fluorine. Liquid fluorine it is most expedient to use paired with such fuels as ammonia, hydrazine, pentaborane and especially liquid hydrogen. fuel/propellant F_2+H_2 exceeds O_2+H_2 to 4-5o/o on the mass specific impulse, developed with engine, and on 70o/o on density specific impulse, and also on the density (to 55o/o). It is most useful for ZhRD of upper booster stages and for ZhRD of space vehicles with the relatively short flight time and large necessary total impulse/momentum/pulse [1].

Fuel shortages F_2+H_2 include:

1) the high temperature of combustion products, which complicates cooling chamber/camera;

2) the high cost/value of fluorine;

3) the large toxicity of fluorine and combustion products (HF).

The high values of the density specific impulse, developed with engine during the utilization of fuels/propellants on the basis of fluorine, can be judged from given below table 10.7.

For ZhRD of space vehicles is possible the use/application of the fuels/propellants: F_2+NH_3, $F_2+N_2H_4$, F_2+MMG, F_2+CH_4, $F_2+B_2H_6$, etc.

Fuels/propellants on the basis of the fluorine-bearing oxidizers. Oxide of fluorine OF_2 it is expedient to use paired with liquid hydrogen, UDMH, MMG, hydrazine, ammonia and methane. For ZhRD of space vehicles is possible the utilization of fuel/propellant $OF_2+B_2H_6$.

Table 10.6. Characteristics of the inflammation of some fuels/propellants.

(1) Горючее	(2) Окислитель					
	O_2	H_2O_2	HNO_3	N_2O_4	F_2	ClF_3
H_2	Н	Н	Н	Н	С	С
N_2H_4	Н	К	С	С	С	С
NH_3	Н	Н	К	К	С	С
ММГ	Н	Н	С	С	С	С
НДМГ	Н	Н	С	С	С	С
C_2H_5OH	Н	Н	Н	Н	С	С

Note. N - incombustible fuels/propellants; S - hypergolic fuels; K - fuels/propellants, which ignite spontaneously in the presence of catalyst.

Key: (1). Oxidizer. (1). Combustible.

Page 155.

The mass and density specific impulses of the engines, which use fuels/propellants on the basis of fluorine oxide, are more than the analogous parameters of ZhRD, which work on the fuels/propellants on the basis of liquid oxygen. All fuels/propellants on the basis of fluorine oxide are igniting spontaneously and possess, with exception of fuel/propellant OF_2+H_2, a comparatively high density.

Due to the high cost/value of fluorine oxide in a number of cases it is expedient to use mixture F_2+O_2, which is only a little inferior to it on the effectiveness.

As the igniting spontaneously storable propellants very advanced propellants $ClF_3+N_2H_4$ and especially $ClF_5+N_2H_4$. One of the difficulties, which appear during the utilization of fuel/propellant $ClF_5+N_2H_4$, is the formation of solid precipitation on internal surface of chamber wall.

Highly efficient is fuel/propellant $BrF_5+B_5H_9$; its density is equal to 1990 kg/m³. Volumetric jet firing, which works on similar of fuel/propellant, when $p_к$=68.7 bar [70 kgf/cm²] and p_c=0.981 bar [1 kgf/cm²], equilibrium expansion and optimal coefficient $\varkappa$ is equal to 4.81 N•s/m³ [489 kgf•s/l].

To a number of effective ones relate the fuels/propellants on the basis of trifluoride of nitrogen NF_3 and especially tetrafluorinehydrazine N_2F_4 with the utilization of hydrazine, pentaborane and liquid hydrogen as the fuels. However, the use/application of tetrafluorinehydrazine impedes its high cost/value.

§10.6. Selection of the optimal excess oxidant ratio $\alpha_{ок}$

After the selection of components is designed the optimum value of the fuel component ratio $\varkappa$ or coefficient $\alpha_{ок}$. During calculations indicated find maximum expressions $I_{уд}hQ^{c}_{т}$.

Table 10.7. The density specific impulse of oxygen and fluoride ZhRD at the level of sea ($p_к$=66,7 bar [68 kgf/cm²]; p_c=0,981 bar [1 kgf/cm²]; $x=x_{опт}$; equilibrium expansion).

(2) Горючее	(1) Окислитель			
	O_2		F_2	
	$I_{уд.об.з}$			
	(3) $\frac{н \cdot сек}{м^3}$	(4) $\frac{кГ \cdot сек}{л}$	(3) $\frac{н \cdot сек}{м^3}$	(4) $\frac{кГ \cdot сек}{л}$
H_2	1,069	109	1,834	187
NH_3	2,569	262	4,119	420
N_2H_4	3,258	335	4,678	477

Key: (1). Oxidizer. (2). Combustible. (3). N•s/m³. (4). kgf•s/l.

Page 156.

Specific jet firing and density of fuel/propellant depend on coefficient $\alpha_{ок}$, i.e. $I_{уд\,h}=f(\alpha_{ок})$ and $\varrho_т=f(\alpha_{ок})$.

Specific impulse has the great value with the excess of fuel, i.e., when $\alpha_{ок}<1$, but not with the stoichiometric relationship/ratio of propellant components, in connection with the fact that with the excess of fuel in composition of combustion products is increased the content of gases with the low molecular weight (CO, H_2, etc.) in comparison with the content of gases with the increased molecular weight (CO_2, H_2O, etc.). Furthermore, when $\alpha_{ок}<1$ descends the

temperature of combustion products, which leads to the decrease of the expenditures of the chemical energy of fuel/propellant for dissociation. Simultaneously is facilitated cooling the chamber/camera: by high fuel consumption to more easily cool the chamber/camera in which moreover, the combustion products have a reduced temperature.

With an increase of the temperature of combustion products the value of coefficient $\alpha_{ок.}$ with which is provided the maximum of specific impulse, it is decreased.

Usually the density of oxidizer is more than the density of fuel, i.e., $\varrho_{ок} > \varrho_{г}$. Therefore with the decrease of coefficients $\alpha_{ок}$ and $\varkappa$ the density of fuel/propellant $\varrho_{т}$ is decreased.

As a result of the density effect of fuel/propellant the value of coefficient $\alpha_{ок.}$ at which reaches the maximum of the characteristic velocity of rocket vehicle, is displaced from the value of coefficient $\alpha_{ок.}$ corresponding to maximum specific impulse, to the side of smaller values.

The optimum values of coefficients $\alpha_{ок}$ and $\varkappa$ depend also on the expansion ratio of gas ε_c. For the chamber/camera with the high expansion ratio of gas $\alpha_{ок.опт} \to 1$ (usually $\alpha_{ок.опт} = 0.95 = 0.98$) as a

result of the more complete course of the reactions of recombination. Table 10.8 gives the values of coefficient $\varkappa$ used for some fuels/propellants of ZhRD.

Table 10.8. Values of coefficient κ for some fuels/propellants, utilized in ZhRD.

(1) Топливо		κ
(2) окислитель	(3) горючее	
O_2	H_2	4,50—5,50
O_2	(4) Керосин	2,20—2,38
O_2	NH_3	1,25
F_2	H_2	8,00—12,00
N_2O_4	$H_2N-NH(CH_3)$	1,64—2,54
N_2O_4	(5) Аэрозин-50	1,50—2,00
85%-ная (6) H_2O_2	(4) Керосин	8,2

Key: (1). Fuel/propellant. (2). oxidizer. (3). combustible. (4). Kerosene. (5). Aerozine.

Page 157.

§10.7. One-component liquid propellants.

most widely as the monopropellant ZhRD are used peroxide of hydrogen and hydrazine, which are the substances, capable of being decomposed/expanded in the presence of catalyst with the liberation of heat.

The highly concentrated (90-98c/o) aqueous solution of peroxide of hydrogen during the utilization as the monopropellant provides

specific impulse of ZhRD 1500-1900 N•s/kg [≈150-190 kgf•s/kg], in this case the temperature of steam gas in the chamber/camera of decomposition composes 875-1250°K. During the large expansion of the decomposition products of peroxide of hydrogen in the nozzle of chamber/camera water vapors are condensed, which causes certain reduction/descent in the specific jet firing.

Hydrazine is more effective monopropellant than peroxide of hydrogen. It is decomposed/expanded during the heating to 750°K, forming the gaseous products NH_3, H_2 and N_2 (during the complete decomposition only H_2 and N_2).

The decomposition products of hydrazine have sufficiently high temperature (to 1475°K), low molecular weight and they are not inclined to the condensation. Hydrazine provides obtaining specific jet firing 2200-2400 n•s/kg [≈220-240 kgf•s/kg].

One-component ZhRD, which work on peroxide of hydrogen or hydrazine, possess smaller specific impulse, but the reliability of the operation their higher than two-component ones. Therefore peroxide of hydrogen or hydrazine usually is used as the fuel/propellant for auxiliary ZhRD with the low thrusts, including for the satellites and hydrazine braking ZhRD with multiplying and controllable thrust for the soft landing on Mars of KA,

developed/processed in the USA [1].

With the addition of nitric acid or nitrate of hydrazine $N_2H_5NO_3$ (component with the oxidative properties) into hydrazine is raised specific jet firing and density of fuel/propellant, and also descends freezing point (for example, to 253°K [-20°C] with addition 24o/o $N_2H_5NO_3$ throughout the mass).

The monopropellant, representing mixture 75o/o N_2H_4, 24o/o $N_2H_5NO_3$ and 1o/o H_2O, provides specific jet firing to 2600 n•s/kg [≈260 kgf•s/kg] and the density of 1110 kg/m³, i.e., in the energy characteristics it approaches average-energy bipropellants of the type N_2O_4+aerozine-50.

Table 10.9 gives values $T_к$ and $I_{уд.п}$ for ZhRD, which work on different monopropellants [1].

In one-component ZhRD are possible the utilization of other fuels/propellants, including of ammonia, UDMH, isopropylnitrate $(CH_3)_2CHONO_2$, etc.

The products of decomposition of hydrazine, peroxides of hydrogen and UDMH use also as the gaseous working medium/propellant of turbine in two-component ZhRD with the pump feed.

Page 158.

§10.8. The combustible and three-component fuels/propellants containing metal.

One of the directions of an increase in the specific impulse of ZhRD and the density of fuel/propellant is the utilization of metals (Li, Be, Al, Mg), their hydrides (LiH, BeH_2, etc.), and also boron. It is possible to use them:

a) in the form of suspension or colloidal solution of metal in the fuel;

b) in the form of the third component, stored in the separate tank and supplied to the chamber/camera on the separate main.

For each fuel/propellant should be selected the type of metal and its optimal content. For example, into the fuels/propellants O_2+H_2 it is expedient to add beryllium, and into fuel/propellant F_2+H_2 - lithium. The relationship/ratio of the constituent elements of fuels/propellants O_2+Be+H_2 and F_2+Li+H_2 is expedient to select by such so that the chemical reaction would occur between the oxidizer

(O_2, F_2) and the metal (Be, Li), and hydrogen was used as the inert working body, which lowers the molecular weight of the gases, which escape from the nozzle.

Instead of the fuel/propellant F_2+Li+H_2 it is possible to use fuels/propellants F_2+LiH+H_2; lithium hydride vaporizes better than lithium.

After the development of the methods of stabilization of liquid ozone by most powerful/thickest chemical fuel/propellant it will be, probably, fuel/propellant O_3+Be+H_2.

The use/application of fuels/propellants with the fuel containing metal impedes the following.

Table 10.9. Some characteristics of ZhRD, that work on the monopropellants.

Топливо (1)	Верхнее значение температуры термической стабильности (2)	$T_к$	$I_{уд.п}$ при (3) $p_к$=9,81 бар (4) [10 кГ/см²] (5)	
		°K	н·сек/кг (6)	кГ·сек/кг (7)
80%-ная H_2O_2	—	1150	1765	180
98%-ная H_2O_2	383	1240*[1]	1893*	193*
N_2H_4	533	1345**[2]	2422**[2]	247**[2]
N_2H_4 (75%)+$N_2H_5NO_3$ (24%) +H_2O (1%)	491	1615**[2]	2569**[2]	262**[2]

Key: (1). Fuel/propellant. (2). Upper value of temperature of thermal stability. (3). with. (4). bar. (5). kgf/cm². (6). n•s/kg. (7). kgf•s/kg.

FOOTNOTE 1. The values are given taking into account to the condensation of water vapors during the motion along the nozzle.

2. Values are given for condition that 40o/o of formed ammonia are decomposed/expanded into nitrogen and hydrogen. ENDFOOTNOTE.

Page 159.

1. Difficulties of mixing of powdered metal and liquid fuels, especially cryogenic.

2. metallic particles with transport and in process of prolonged storage are settled. It is possible to mix fuel with the powdered metal directly on the launching site; however, in this case appear large operational inconveniences. Particles are settled to a lesser degree with an increase in viscosity/ductility/toughness and density of fuel (for example, with the addition of wax or paraffin), and also with the decrease of the sizes of the particles of the metal (to 1-40 μm).

3. Production of powdered metal, which especially contain beryllium (i.e. powders Be and BeH_2), is characterized by complexity and high costs. Furthermore, powders Be and BeH_2 it possesses high toxicity, which excludes the possibility of their use/application as the additions to the fuel for the first-stage engines of carrier rockets.

4. During the supplying of fuel containing metal into chamber/camera is possible soiling injectors. Definite difficulties causes the organization of the combustion of metallic particles.

During the utilization of a three-component fuel/propellant the metal can be supplied into the chamber/camera in the molten form, for example, by the compressed inert gas. During testing of the experimental chamber/camera, which works on the three-component of fuel/propellant F_2+Li+H_2 with content of 10-12o/o Li, is obtained the specific impulse in the vacuum of more than 5000 n•s/kg [≈500 kgf•s/kg [1]. Metal can be introduced into the chamber/camera on the separate main and it is direct in the form of finely dispersed powder, which however, is connected with the great difficulties.

The shortcomings of ZhRD, which use a three-component fuel/propellant, include the complexity of their construction/design and changing the operating mode.

The use/application of the combustible and three-component fuels/propellants containing metal leads to an increase of the heat fluxes into the chamber walls, which complicates its cooling and raises requirements for the structural materials. ZhRD, which work on such fuels/propellants, it is most expedient to use for the space vehicles and the latter/last booster stages.

Page 160.

Chapter XI.

HEAT TRANSFER AND COOLING OF ZhRD.

§11.1. Transmission modes of heat fluxes.

With the work of the majority of the rocket engines of the wall of their chambers/cameras they receive a significant quantity of heat from the products of combustion or decomposition of the components of propellant or products of heating working medium/propellant. For the safeguard of reliable chamber operation and engine as a whole it is necessary to abstract/remove the heat indicated to those or in another manner.

The transfer of the heat (it they call also heat transfer) quantitatively determines the values of heat flux and heat transfer rate.

Heat flux is called the energy content, transmitted in the form of heat per unit time through any surface of F. Heat flux measures in the watts [J/s; kcal/s] and they designate by letter Q.

Heat transfer rate (or heat-flux density) is called the heat flux, per unit surface area. heat transfer rate characterizes heat-transfer intensity; it they designate by letter q. Consequently,

$$q=\frac{Q}{F}. \qquad (11.1)$$

Heat transfer rate has a dimension W/m² [J/(s•m²); kcal/h•m²].

Most heat-stressed is the area of nozzle throat. For some types of ZhRD heat transfer rate in the cross section indicated reaches 70•10⁶ W/m² [60.2•10⁶ kcal/h•m²)].

Heat fluxes can be transmitted by convection, thermal radiation and thermal conductivity of medium (substance)[1].

FOOTNOTE [1]. For greater detail, see [17] and [18]. ENDFOOTNOTE.

Specific convection current designate $q_{кон}$, and radiant (or radiation) - $q_{л}$.

The relative value of convective and radiant fluxes in different types of rocket engines is to a considerable extent dissimilar.

In the chambers/cameras of ZhRD with the coolant passage the

basic form of heat transfer are convective heat fluxes from the combustion products to the internal chamber wall and from it to the coolant (propellant component).

Heat transfer by thermal radiation in ZhRD has noticeably smaller value. However, the nozzles of the chambers/cameras of some ZhRD in the finite segment do not have the coolant passage. The thermal condition of this section (nozzle) of nozzle is determined by heat transfer from the combustion products to the nozzle liner and by heat transfer by radiation from the wall into the surrounding space and to the combustion products.

For YaRD and thermal ERD are characteristic the higher temperatures of gas, than for the chemical engines. Therefore the role of radiation/emission in YaRD and ETRD noticeably increases/grows. Furthermore, in YaRD the convective heat exchange makes the complex problem of the branch/removal of heat from the fuel elements to the working medium/propellant.

Page 161.

Electrical rocket engines are characterized by the very low pressure of plasma in their chamber/camera. Therefore the convective heat fluxes, which depend on the gas pressure in the chamber/camera,

are also low and the thermal condition of these engines is determined in essence by radiant fluxes from the plasma to the chamber walls and from them into outer space.

§11.2. Heat transfer by convection path.

Let us examine the equations according to which it is possible to determine specific convection current from the gas to the surface of wall and from the wall to the coolant.

Let us introduce the following designations of the parameters of chamber/camera with the coolant passage:

$T_{газ.в}$ - the temperature of the reduction of gas, which is determining heat transfer from the gas to the wall;

$T_{в.п}$ - temperature of the heated (receiving heat) surface of internal chamber wall;

$T_{о.п}$ - temperature of the cooled (giving up heat) surface of internal chamber wall;

$T_{ох}$ - temperature of the coolant, which takes place through the coolant passage;

$q_{кон.н}$ - specific convective heat flux, transmitted from the gas to is heated the surface of internal wall;

$q_{о.п}$ - specific convective heat flux, transmitted from the cooled surface of internal wall to the coolant;

$\alpha_{н.п}$ - coefficient of convective heat emission from the gas to the heated surface of internal wall;

$\alpha_{о.п}$ - coefficient is convective heat transfer from the cooled surface of internal wall to the coolant.

The coefficient of convective heat emission expresses a quantity of heat, transmitted by convection path through unity of surface per unit time to each degree of a difference in the temperatures of wall and gas or liquid; this coefficient has a dimension W/(m²•deg)[kcal/(h•m²•deg)].

Therefore specific flow $q_{кон.н}$ is designed from the equation

$$q_{кон.н}=\alpha_{н.п}(T_{газ.в}-T_{н.п}), \qquad (11.2)$$

but specific flow $q_{о.п}$ - according to the equation

$$q_{о.п}=\alpha_{о.п}(T_{о.п}-T_{ож}). \qquad (11.3)$$

Temperature $T_{\text{газ.п}}$ somewhat less than the temperature of stagnation of gas, since the part of the heat, which was isolated during braking of gas in the boundary layer, is abstracted/removed from it by convection path, also, because of the thermal conductivity.

All difficulties of calculating the convective heat exchange are reduced to the determination of heat-transfer coefficients $\alpha_{\text{к.п}}$ and $\alpha_{\text{л.п}}$.

Page 162.

They are designed, using a dependence between the dimensionless criteria - the criteria of Nusselt, Reynolds and Prandtl:

$$\mathrm{Nu}=f(\mathrm{Re},\ \mathrm{Pr}). \qquad (11.4)$$

The criteria indicated determine the character of a change in the rate and temperature in the boundary layer which to a considerable degree influences the convective heat exchange.

Utilization of dependence (11.4) for the case of heat exchange between the products of the combustion of chamber/camera of ZhRD and its wall gives the following formula for calculating the heat-transfer coefficient $\alpha_{\text{к.п}}$:

$$\alpha_{\text{к.п}}=B_1 s\frac{\dot{m}^{0.8}}{d^{1.8}}, \qquad (11.5)$$

where B_1 - combination of the thermophysical properties of combustion products, depending on their composition and temperature; σ - dimensionless coefficient, considering the effect of a change in temperature and Mach number on the height/altitude of boundary layer; $\dot{m}$ - mass flow rate per second of combustion products; d - diameter of chamber/camera.

Coefficient $\alpha_{к.п}$ depends on product ρW and increases/grows with its increase. This is explained by the fact that with an increase in gas density is increased a quantity of fractions/particles of gas per unit of volume, and with an increase in the gas velocity increases/grows a quantity of its particles, which pass per unit time in wall. During the convective heat exchange the heat is transferred by particles. Therefore with an increase in the density of gas and its rate the process of heat emission from the gas to the wall becomes more intensively, i.e., values $\alpha_{к.п}$ and $q_{кон.п}$ increase/grow. Product ρW has maximum value in the critical cross section (see §4.5); consequently, and values $\alpha_{к.п}$ and $q_{кон.п}$ also are maximum in this cross section (Fig. 11.1).

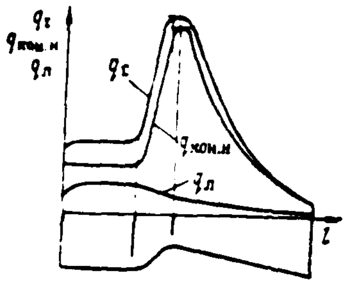

Fig. 11.1. Graphs of the distribution of specific fuel flows q_Σ, $q_{кон.н}$ and $q_л$ along the length of chamber/camera.

Page 163.

Utilization of dependence (11.4) in connection with heat exchange between the wall and coolant at the high values of the heat fluxes, characteristic for the chambers/cameras of ZhRD, gives the following formula for calculating the coefficient of heat emission $\alpha_{о.ж}$:

$$\alpha_{о.ж} = B_2 \beta \left(\frac{\dot{m}_{ох}}{f_{о.т}}\right)^{0.8} \frac{1}{d_{тэх}^{0.2}}, \qquad (11.6)$$

or taking into account equation (4.9)

$$\alpha_{о.ж} = B_2 \beta (\varrho_{ох} W_{ох})^{0.8} \frac{1}{d_{тэх}^{0.2}}, \qquad (11.7)$$

where B_2 - combination of the thermophysical properties of coolant, depending on the type of coolant and its temperature: β - coefficient, which calculates a change in the thermophysical properties of coolant on the basis of the height/altitude of boundary

layer: $\dot{m}_{ox}$, ρ_{ox} and W_{ox} - mass flow rate per second, density and the rate of coolant respectively; $f_{o.т}$ - area of cross section of the coolant passage; $d_{гид}$ - the hydraulic (equivalent) diameter of the coolant passage, determined according to the equation

$$d_{гид}=\frac{4f_{о.т}}{l_i},$$

let us accept P - the complete (moistened) girth of the coolant passage.

§11.3. Heat transfer by radiation.

Solid bodies radiate the waves of all lengths from λ=0 to λ=∞, i.e., their radiation/emission is characterized by continuous spectrum.

Gases radiate and absorb electromagnetic energy only in the specific wavelength ranges Δλ, i.e., radiation/emission and gas absorption is characterized by the so-called line spectrum. This radiation/emission and absorption are called selective, or selective. The simpler the structure of molecule or atom, the more brightly expressed the linear structure of radiation spectrum, and the necessary the account of this structure of spectrum during calculation of radiation/emission.

Selective radiation is entirely specific to the working

medium/propellants of electrical rocket engines, i.e., to monatomic gases - to cesium, to lithium, to argon and so forth; calculation of their radiation/emission is very difficult. However, carried out calculations show sharp increase of radiant fluxes q_n with an increase in the temperature of gases.

From the gases, entering composition of combustion products of chemical fuels/propellants, to the greatest degree radiate and absorb energy the polyatomic gases, which have the unsymmetric structure of molecule, first of all the water vapor H_2O and carbon dioxide CO_2. the radiating and absorptive power of mono- and diatomic gases can be disregarded/neglected.

Solid bodies usually radiate and absorb energy by surface, and gases - by entire volume. Therefore the radiating and absorptive power of the gases, which contain H_2O and CO_2, determines not only the temperature of gas and partial pressures H_2O and CO_2, but also the combustion chamber configuration; the latter, in turn, characterizes the mean pathlength of ray/beam l.

Page 164.

The radiation/emission of water vapor and carbon dioxide is subordinated with some assumptions to the Stefan-Boltzmann law; for

calculating the radiant thermal flux from them to the chamber wall it is possible to use the equation

$$q_{\text{л}} = \varepsilon_{\text{ст.э}}\left[\varepsilon_{\text{газ}} c_0 \left(\frac{T_{\text{газ}}}{100}\right)^4 - \varepsilon'_{\text{газ}} c_0 \left(\frac{T_{\text{н.п}}}{100}\right)^4\right], \qquad (11.8)$$

where $\varepsilon_{\text{ст.э}}$ - effective emissivity factor of the heated surface of internal chamber wall;

$\varepsilon_{\text{газ}}$ and $\varepsilon'_{\text{газ}}$ - emissivity factor of gas respectively at temperatures $T_{\text{газ}}$ and $T_{\text{н.п}}$;

c_0 - radiation coefficient of absolutely black solid body, equal to 5.67 W/(m²•deg)⁴) [4.96 kcal/(h•m²•deg⁴)].

Value $\varepsilon_{\text{ст.э}}$ depends on emissivity factor of wall and gas ($\varepsilon_{\text{ст}}$ and $\varepsilon_{\text{газ}}$ respectively). Values $\varepsilon_{\text{ст}}$, definable by material of wall and by the state of its heated surface, take from table [17].

Value $\varepsilon_{\text{газ}}$ for the combustion products, which contain water vapor and carbon dioxide, is equal to

$$\varepsilon_{\text{газ}} = \varepsilon_{H_2O} + \varepsilon_{CO_2} - \varepsilon_{H_2O}\varepsilon_{CO_2}. \qquad (11.9)$$

The presence of latter/last term in equation (11.9) is explained by partial mutual radiation absorption H_2O and CO_2.

Values ε_{H_2O} and ε_{CO_2} depend on the temperature of gas and on the product of its partial pressure on the mean pathlength of ray/beam l,

while value ε_{H_2O}, furthermore, from the pressure of combustion products p_{Σ}. For determination ε_{H_2O} and ε_{CO_2} are used special graphs [17].

The distribution of specific radiant thermal fluxes $q_{л}$ along the length of chamber/camera is shown in Fig. 11.1; they are maximum in the combustion chamber, since in it a pressure (and, consequently, value p_{H_2O} and p_{CO_2}) and temperature $T_{газ}$ have the greatest values.

Taking into account the degree of approximation of calculations of radiation/emission, one should determine value $q_{л}$ only for the flow core in the combustion chamber (the value indicated let us designate $q_{л.к}$), and value $q_{л}$ in other cross sections to accept the following:

1) it is direct in the fire bottom of the head

$$q_{л}=0{,}8\ q_{л.к};$$

2) in the section, removed at a distance of 50-100 mm from the fire bottom of head to the cross section of the tapering portion of the nozzle with a diameter of $d=1{,}2d_{кр}$, value $q_{л}$ is constant and equal to $q_{л.к}$.

3) in the critical cross section

$$q_{л}=0{,}5q_{л.к};$$

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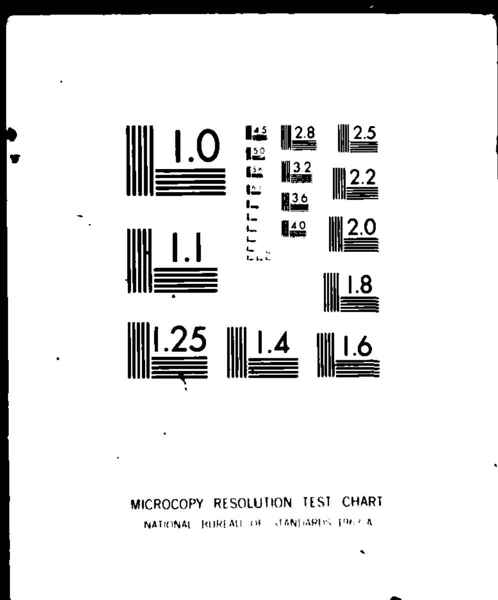


4) in the expanding section of nozzle with a diameter of $d=1{,}5d_{кр}$ $q_л=0{,}1q_{л.к}$;

5) the subsequent cross section of nozzle with diameter $d=2{,}5d_{кр}$ $q_л=0{,}02q_{л.к}$.

Page 165.

After connecting the points of smooth curve indicated, we will obtain the distribution of the radiant thermal flux along the chamber/camera.

With temperature of the combustion products of 2000°K $q_л$ it is small in comparison with specific convection current $q_{конв}$, but at a temperature of combustion products of 3000-4000°K value $q_л$ can reach 30o/o of the general/common/total heat flux in the wall.

§11.4. Heat transfer as a result of the thermal conductivity of material of wall.

If with the work of engine with those or by another method to provide a difference in the temperatures of the surfaces of chamber wall, the heat through the wall is transmitted because of the thermal conductivity of its material. In this case heat transfer rate is

determined according to the equation

$$q_{ст}=\frac{\lambda_{ст}}{\delta_{ст}}(T_{г.п}-T_{о.ж}), \quad (11.10)$$

where $\delta_{ст}$ - wall thickness; $\lambda_{ст}$ - coefficient of the thermal conductivity of material of wall, which characterizes its ability to conduct heat.

With the same thickness of wall $\delta_{ст}$ for the safeguard of prescribed/assigned flow $q_{ст}$ through the wall the necessary difference in the temperatures of wall the less, the greater the coefficient $\lambda_{ст}$. On the contrary, with low coefficient $\lambda_{ст}$ a difference in the temperatures of wall $T_{г.п}-T_{о.ж}$ can be sufficient large on the thin chamber wall. For example, with moderate heat transfer rate through wall $q_{ст}=11.6\cdot10^6$ W/m² [$10\cdot10^6$ kcal/(h·m²)] a difference in the temperatures on the wall with a thickness of 1 mm made of the stainless steel is equal to

$$T_{г.п}-T_{о.ж}=\frac{q_{ст}\delta_{ст}}{\lambda_{ст}}=\frac{11.6\cdot10^6\cdot1\cdot10^{-3}}{23.3}=500 \overset{(1)}{\text{град}}.$$

Key: (1). deg.

Of all materials, with exception of noble metals, pure copper possesses the greatest coefficient of thermal conductivity. For copper, impure, and the alloys of copper with other metals (for example, bronze of one or the other composition) of value λ it is noticeably less.

The coefficient of the thermal conductivity of the metals and other materials depends on their temperature. Fig. 11.2 depicts graphs $\lambda=f(T)$ for pure copper and stainless steel.

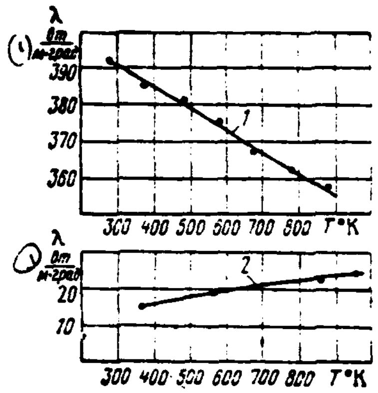


Fig. 11.2. Dependence is the coefficient of thermal conductivity for pure copper (1) and stainless steel (2) on the temperature.

Key: (1). W/m•deg.

Page 166.

Taking into account the dependence indicated, the coefficient of thermal conductivity in equation (11.10) must be taken at mean temperature of the wall

$$T_{\text{ст.ср}}=\frac{T_{\text{г.с}}+T_{\text{ж.с}}}{2}.$$

§11.5. Characteristic of heat transfer through the chamber wall.

In the process of the work of the engine of chamber wall receive both convective and radiant thermal fluxes. Therefore total heat

transfer rate, which enters the chamber walls, is equal to

$$q_{\Sigma}=q_{н.п}=q_{конв.п}+q_{л}. \qquad (11.11)$$

Heat transfer rate $q_{н.п}$ can be written also in the following form:

$$q_{н.п}=\alpha'_{н.п}(T_{газ.п}-T_{н.п}), \qquad (11.12)$$

where $\alpha'_{н.п}$ - certain effective heat-transfer coefficient, considering both convective and radiation heat exchange between the combustion products and the wall.

On the basis of equations (11.2), (11.11) and (11.12) the heat-transfer coefficient $\alpha'_{н.п}$ is equal to

$$\alpha'_{н.п}=\alpha_{к.п}+\frac{q_{л}}{T_{газ.п}-T_{н.п}}. \qquad (11.13)$$

The graph of the distribution of total heat transfer rate q_{Σ} along the length of chamber/camera is depicted in Fig. 11.1. Due to the effect of radiant flux the maximum of total heat transfer rate somewhat will be mixed from the critical cross section to the side of the head of chamber/camera. From the graph, depicted in Fig. 11.1, it follows that the area of critical cross section is the most heat-stressed section of chamber/camera; therefore its reliable cooling causes the greatest difficulties.

The heat fluxes, which come from combustion products the wall, pass through it and they are transmitted to the coolant, which takes place through the coolant passage.

In the beginning of the work cf engine tc ccclant is transmitted not entire heat flux, which ccmes the wall frcm combustion products, but cnly its part; another part gces to the warm-up cf chamber walls, as a result of which the temperature of chamber wall ccntinucusly increases/grows. With an increase cf the temperature of wall that part of the heat flux which is spent on the warm-up of walls, continuously it is decreased. Consequently, the initial operating cycle of engine is characterized by nonstationary system of cooling. If they are made the specified ccnditicns, then after certain period of the time (for the chamber/camera of ZhRD it is low) is established/installed equilitrium.

Page 167.

It is characterized by the fact that entire heat flux, which comes the wall frcm combustion products, is transmitted frcm the wall to the ccolant. Therefore, if we ccnsider that the areas of that heated and cooled the surface cf wall are equal to each other, then is provided the equality

$$q_{\text{в.к}} = q_{\text{ст}} = q_{\text{о.ж}} = \text{const.}$$

In entire section cf the transfer of heat flux - from the boundary layer of combusticn prcducts to the boundary layer cf

coolant - is established/installed the constant distribution of temperatures (Fig. 11.3), sc that, in spite of the presence cf heat flux, temperature $T_{н.ж}, T_{о.ж}$ and $T_{ох}$ they remain constants. Consequently, in this case is provided steady state cf cooling chamber/camera. The heat flux indicated during steady state cf cooling we will subsequently aralcgcus with equation (11.11) designate q_{Σ}. Consequently,

$$q_{н.ж}=q_{ст}=q_{о.ж}=q_{\Sigma}.$$

Therefore equations (11.12), (11.10) and (11.3) can be written in the fcllcwing form:

$$q_{\Sigma}=\alpha'_{н.ж}(T_{газ.н}-T_{н.ж});$$

$$q_{\Sigma}=\frac{\lambda_{ст}}{\delta_{ст}}(T_{н.ж}-T_{о.ж});$$

$$q_{\Sigma}=\alpha_{о.ж}(T_{о.ж}-T_{ох}).$$

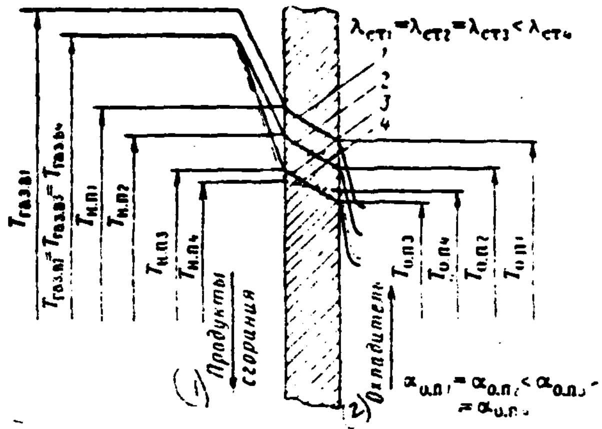


Fig. 11.3. Graphs of distribution of the temperature of internal wall on its thickness and in near-wall layers in different parameters of gas, coolant and internal wall.

Key: (1). Combustion products. (2). Coolant.

Page 168.

Hence

$$T_{н.п}=T_{газ.п}-\frac{q_{\Sigma}}{\alpha'_{г.п}}; \quad (11.14)$$

$$T_{н.п}=T_{о.п}+\frac{q_{\Sigma}\delta_{ст}}{\lambda_{ст}}; \quad (11.15)$$

$$T_{о.п}=T_{ох}+\frac{q_{\Sigma}}{\alpha_{о.п}}. \quad (11.16)$$

After substituting expression (11.16) in equation (11.15), we will

obtain

$$T_{\text{н.п}}=T_{\text{ох}}+\frac{q_{\Sigma}}{\alpha_{\text{о.п}}}+\frac{q_{\Sigma}\delta_{\text{ст}}}{\lambda_{\text{ст}}}. \quad (11.17)$$

Equations examined above are conveniently used for the analysis of the effect of different parameters on the mode/conditions of cooling chamber/camera. A change in any of the parameters indicated causes to a certain degree a change in the graph of the temperature distribution in a wall layer from the side of combustion products, according to wall thickness and in a wall layer from the side of the coolant (see Fig. 11.3).

For example, if is increased the temperature of combustion products $T_{\text{газ.в}}$, then temperatures $T_{\text{н.п}}$ and $T_{\text{о.п}}$ increase/grow (see curves 1 and 2), moreover simultaneously increases/grows temperature $T_{\text{ох}}$. During the replacement of material of wall to the material with the high coefficient of thermal conductivity $\lambda_{\text{ст}}$ descends temperature $T_{\text{н.п}}$, but somewhat increases/grows temperature $T_{\text{о.п}}$ (see curves 3 and 4). The same effect is observed with the decrease of the thickness of internal wall $\delta_{\text{ст}}$. If we in some manner or other (for example, by an increase in the rate of coolant $W_{\text{ох}}$) increase the branch/removal of heat fluxes from the wall to the coolant, then simultaneously they descend temperature $T_{\text{н.п}}$ and $T_{\text{о.п}}$ (see curves 2 and 3).

As can be seen from equations (11.14), (11.15) and (11.16), a

difference in temperatures and, consequently, also the angle of slope of the distribution curve of temperature they are decreased:

a) for the wall layer of combustion products, in which their temperature descends from $T_{газ.в}$ to $T_{к.п.}$ - with the decrease of value q_2 and the increase of heat-transfer coefficient $\alpha_{к.п}$;

b) for the wall - with the increase of the coefficient of the thermal conductivity of its material $\lambda_{ст}$;

c) for a wall layer of the coolant in which its temperature drops from $T_{о.ж}$ to $T_{ож}$, - with the decrease of value q_2 and the increase of heat-transfer coefficient $\alpha_{о.ж}$.

The effect of the parameters of the products of combustion, wall, coolant and coolant passage on the mode/conditions of cooling in more detail is examined in §§11.7-11.9.

Page 169.

§11.6. Requirements for the cooling system of engine chamber.

If we do not abstract/remove from the wall heat fluxes, then through certain period of the time of riding-crop it will be

overheated, and will occur the inadmissible reduction/descent in the strength of material from which it is prepared, or its hot spot, which can lead to the decomposition of chamber/camera.

Due to the high rate of products the combustion, especially in the expanding section of nozzle, chamber wall undergo erosion (washout). The erosion of walls becomes especially noticeable during their superheating, since in this case the errosive resistance of material descends. Therefore for the reliable chamber operation of engine the temperature of its wall must not exceed the value, which is the permissible for the selected material of wall according to strength conditions and errosive resistance, i.e.

$$T_{ст} < T_{доп},$$

but for the chamber/camera with the external flowing cooling

$$T_{ж.п} < T_{доп}.$$

the system of external flowing cooling must provide the following additional conditions.

1. Temperature of cooled surface riding-crops $T_{o.п}$ in all sections of chamber/camera must not exceed value at which will begin that called skin boiling; presence of skin boiling leads to considerable decrease of heat flux, abstracted/removed from wall to coolant, and to its hot spot (see pg. 174).

2. Temperature of coolant must not reach such values at which coolant begins to be decomposed/expanded with formation of solid, resinous or gaseous decomposition products. Solid and resinous particles are deposited on the wall, forming a layer with the low coefficient of thermal conductivity. In this case the transfer of heat from the wall to the coolant is decreased, which causes an increase in the temperature of wall, and it can burn down. Furthermore, solid and tarry particles can clog the openings/apertures of injectors of chamber/camera, which is inadmissible.

During superheating of some components of propellant (H_2O_2, N_2O_4, UDMH), utilized as the coolant, can occur the effects, equipollent to explosion.

3. For ZhRD, which work on diagram "liquid-to-liquid", temperature of coolant, which enters from coolant passage in injectors of chamber/camera, must not exceed boiling point of coolant, i.e.

$$T_{\text{ох}} < T_{\text{кип}},$$

moreover value $T_{\text{кип}}$ it is necessary to take for pressure of coolant, which it has at output/yield from coolant passage.

Page 170.

If conditions $T_{ox}<T_{\text{кип}}$ is not observed, then coolant enters injectors in the form of vapor or emulsion. In this case the mode of operation of injectors, designed for atomization of liquid, sharply is disturbed, and is feasible the explosion of chamber/camera. Furthermore, from the sections of chamber wall which will be cooled by the evaporated coolant, sharply decreases the branch/removal of heat, and is feasible their hot spot.

4. Rate of coolant must not be too large. With its increase grows the heat-transfer coefficient $\alpha_{o.ж}$ and descends temperature $T_{в.ж}$ (see equation (11.17)), but simultaneously is increased the required power of the feed system of propellant components into the chamber/camera and, consequently, also the mass of the system indicated.

5. Coolant passage of chamber/camera must be technologically effective, i.e., its sizes/dimensions and form must be such that it would not appear large difficulties during serial manufacture of chambers/cameras.

§11.7. Effect of different factors on the heat flux from the combustion products to the wall.

The temperature of the combustion products exerts a substantial influence on values $q_{конв}$ and $q_л$, calling with its increase their growth, which is evident from equations (11.2) and (11.8). The tendency of a temperature rise in the combustion products in the chamber/camera of ZhRD is caused by use/application of ever more effective fuels/propellants (with the large heating power) for obtaining the high specific impulse. With an increase in the specific impulse descends the necessary propellant component flow, which causes additional difficulties during cooling of chamber/camera, since coolant is one of the propellant components.

Coefficient $\varkappa$ influences heat fluxes through the temperature of combustion products and partly through their composition. With an increase in the deviation of coefficient $\varkappa$ from the stoichiometric value heat flows from the combustion products to the walls descend.

At values $q_{конв}$ and $q_л$ exerts essential effect pressure $p_к$.

The dependance of heat flux $q_{конв}$ on pressure $p_к$ can be shown based on the example of critical cross section, for it in accordance with equation (11.5)

$$\alpha_{г.к.кр} = B_1 \sigma \frac{\dot{m}^{0.8}}{d_{кр}^{1.8}}.$$

After substituting in the latter/last equation value of $\dot{m}$ from equation (4.14) and after considering relationship/ratio $f_{кр}=\pi d_{кр}^2/4$, after some conversions we will obtain

$$\alpha_{кон.н.кр}=B_1' \frac{p_к^{0.8}}{\beta^{0.8} d_{кр}^{0.2}}. \quad (11.18)$$

Page 171.

As can be seen from equation (11.18), value $\alpha_{кон.н.кр.}$ and, consequently, value $q_{кон.н.кр}$ increases/grows with an increase in pressure $p_к$ to degree of 0.8.

Value $q_л$ also increases with pressure rise $p_к$ in connection with an increase in partial pressures p_{CO_2} and p_{H_2O}, which are determining radiation heat exchange. The greater the heat fluxes into the chamber wall, the greater degree to which is heated the coolant, which takes place through the coolant passage. Consequently, transition/transfer to the higher combustion chamber pressures leads to an increase in the difficulties of cooling the chamber/camera. The dependence indicated occurs with the invariable expenditure/consumption of coolant, so that the increase $p_к$ in question is achieved not by an increase in the propellant component flow (and, consequently, the expenditure/consumption of coolant), but

by the reduction in area of critical cross section.

The effect of total heat flux q_{Σ} on the temperature of the heated surface of chamber wall $T_{н.п}$ can be evaluated on the basis of equation (11.14). In proportion to an increase in value q_{Σ} the temperature $T_{н.п}$ descends, and vice versa: with reduction/descent q_{Σ} temperature $T_{н.п}$ increases/grows also in the extreme case when $q_{\Sigma}=0$ (i.e. in the absence of heat flow through the wall) temperature $T_{н.п}$ becomes equal to the temperature of combustion products $T_{газ.в}$. Consequently, with increase q_{Σ} temperature $T_{н.п}$ descends, if we count the temperature of the products of combustion $T_{газ.в}$ of constant.

The effect of rated thrust of chamber/camera on its cooling is connected with the fact that with the decrease of thrust in the directly proportional dependence descends the propellant component flow and, consequently, also the expenditure/consumption of coolant. The area of the surface of chamber/camera, which must be cooled, is decreased to a lesser degree. Therefore the creation of highly efficient ZhRD, which have thrust is less than 500 n [~~50 kgf] and cooled with the aid of the coolant (especially with the long operating time of engine), it is extremely difficult.

Engine power rating influences on cooling chamber/camera on that reason, that the decrease of the thrust of chamber/camera is achieved

by a reduction/descent in the propellant component flow (and, consequently, coolant), in this case respectively descend values $p_к$ and $q_{кон.н}$. Consequently, with the decrease of pressure $p_к$ simultaneously descend values $\dot{m}_{ох}$ and $q_{кон.н}$.

In accordance with equations (11.14) and (4.14)

and

$$q_{кон.н} \sim p_к^{0.8} \text{ и } \dot{m}_{ох} \sim p_к,$$

i.e. with the decrease of pressure $p_к$ the expenditure/consumption of coolant is decreased to the greater degree than convective heat flow. Furthermore, with the decrease of the expenditure/consumption of coolant descend the rate of coolant in that cooling of channel and coefficient $\alpha_{о.ж}$. Therefore with the reduction/descent the thrusts of the chamber/camera of temperature $T_{ж.ж}$ and $T_{о.ж}$ in accordance with equations (11.15) and (11.16) increase/grow.

Page 172.

Consequently, in proportion to a reduction/descent in the thrust of chamber/camera increase/grow the difficulties of its cooling, what is one of the essential shortcomings in the chambers/cameras with the external flowing cooling.

§11.8. Effect of the parameters of internal chamber wall on its cooling.

The temperature effect of the heated surface of chamber wall $T_{г.с}$ on the mode/conditions of cooling. Permissible temperature $T_{г.с}$ is determined by the heat resistance of material of internal chamber wall. The large the temperature $T_{г.с}$ can be allowed, is those less total heat flow q_{Σ} with the same temp[illegible] $T_{газ.к}$ (see equation 11.14), in this case is decreased the [illegible]ssary value of coefficient $\alpha_{о.ж}$.

In §11.7 it was shown that if we accept $T_{г.с}=T_{газ.к}$, that $q_{\Sigma}=0$, i.e., the necessity for wall cooling drops off. However, temperature $T_{газ.к}$ for the majority of ZhRD is great ($T_{газ.к}=2800\text{-}4000°K$). Therefore temperature $T_{г.с}$ must be decreased, abstracting/removing heat fluxes from the chamber wall. Equality the temperature of the products of combustion and wall can be allowed only for the chamber/camera of one-component ZhRD.

Effect of the coefficient of the thermal conductivity of material of internal chamber wall $\lambda_{ст}$ on the mode/conditions of cooling. With an increase of coefficient $\lambda_{ст}$ is decreased a difference in temperatures $T_{г.с}-T_{ж.с}$ in the invariable parameters of the products of combustion and coolant. If we do not change the parameters of coolant, then with an increase in coefficient $\lambda_{ст}$

descends temperature $T_{\text{г.п}}$, which has certain effect on temperature $T_{\text{о.ж}}$. This effect is explained these, that with a reduction/descent in temperature $T_{\text{г.п}}$ somewhat increase/grow heat fluxes $q_{\text{конв}}$ and $q_{\text{л}}$, which leads to an increase in the temperature of coolant, and the latter in accordance with equation (11.16) - to an increase in temperature $T_{\text{о.ж}}$.

If we compare two materials of internal chamber wall, moreover $\lambda_{\text{ст2}}>\lambda_{\text{ст1}}$, then are valid the following relationships/ratios:

$$T_{\text{в.п2}}<T_{\text{в.п1}} \text{ and } T_{\text{о.ж2}}>T_{\text{о.ж1}}.$$

The greater the coefficient $\lambda_{\text{ст}}$, the less the angle of the slope of straight line of the temperature distribution according to internal wall thickness and the less the temperature $T_{\text{в.п}}$ at a prescribed/assigned temperature $T_{\text{о.ж}}$. Therefore it is expedient for the internal chamber wall to select materials with the highest possible coefficient of thermal conductivity $\lambda_{\text{ст}}$. However, it is necessary to consider the following operational constraints of materials with high value $\lambda_{\text{ст}}$.

1. With increase in coefficient $\lambda_{\text{ст}}$ is decreased difference in temperatures $T_{\text{в.п}}-T_{\text{о.ж}}$, which increases danger of superheating cooled surface of internal wall. Therefore in a number of cases the difference in the temperatures indicated it is necessary to artificially increase, which is reached, as it will be shown below, by an increase in the thickness of internal wall $\delta_{\text{ст}}$.

Page 173.

2. Usually materials with larger coefficiant $\lambda_{ст}$ possess smaller heat resistance, i.e., for them it is necessary to select lower temperature $T_{w.n.}$ in connection with which increase/grow difficulties of cooling chamber/camera.

Let us explain the effect indicated based on the example of chambers/cameras with the steel and copper walls.

The permissible temperature for copper (575°K) and bronze (1075°K) is lower than for stainless steel (1475°K). Therefore during the utilization of a copper or bronze wall with the same temperature T_{max} total heat flow q_{Σ} increases/grows, i.e., from the wall it is necessary to abstract/remove to the coolant greater heat flow. Necessary value $\alpha_{o.n}$ for the copper wall is approximately/exemplarily 2.0-2.5 times more than for the wall made of the stainless steel (with the same thickness of wall) [7].

Let us write equation (11.10) in the following form:

$$q_{\Sigma}=\frac{\lambda_{ст}\Delta T}{\delta_{ст}}. \qquad (11.19)$$

As can be seen from equation (11.19), with the same thickness

δ_{CT} of riding-crop it can pass the greater heat flow, the greater the product $\lambda_{CT}\Delta T$. Calculations show that with the same thickness the wall of copper or bronze is capable of passing 2.5-3.0 times more heat fluxes, than wall made of the stainless steel. Therefore during the intensive cooling of chamber/camera with the copper wall is permitted the elevated temperature of a wall layer.

Effect of the thickness of internal chamber wall δ_{CT} on the mode/conditions of cooling. The decrease of thickness riding-crops δ_{CT} affects heat transfer just as an increase in the coefficient of thermal conductivity λ_{CT}. In accordance with equation (11.19) with the decrease of thickness δ_{CT} increases/grows the heat flux, which wall can pass with the same difference in temperatures $T_{\text{к.п}}-T_{\text{о.п}}$.

The optimum value to which it is expedient to attenuate of wall, depends on total heat flux q_{Σ}. With the increase of value q_{Σ} descends temperature $T_{\text{о.п}}$. Therefore in the area of critical cross section in which heat fluxes have maximum value, wall thickness is selected smallest, but it must provide the required strength of chamber/camera and be technologically feasible.

The temperature effect of the cooled surface of wall $T_{\text{о.п}}$ on the mode/conditions of cooling. With the decrease of total heat flux q_{Σ} increases/grows temperature $T_{\text{о.п}}$, approaching temperature T_{max}.

moreover together with value $T_{\text{н.п}}$ respectively increases/grows value $T_{\text{о.п}}$. But value $T_{\text{о.п}}$ is limited by the temperature of coolant. Temperature $T_{\text{о.п}}$ can exceed the temperature of coolant only to certain permissible value.

Page 174.

Otherwise can begin the decomposition or boiling of coolant. Therefore at low heat flows q_{Σ} and large temperature $T_{\text{н.п}}$ wall thickness it is necessary to increase for obtaining the permissible temperature $T_{\text{о.п}}$.

In order to exclude boiling coolant on the cooled surface of internal chamber wall, temperature $T_{\text{о.п}}$ must be lower than boiling point of coolant at this pressure.

However, in the majority of the cases for the purpose of the intensification of external flowing cooling is provided temperature excess $T_{\text{о.п}}$ above the boiling point of coolant on 10-55 deg, which leads to the simmering of coolant on the cooled surface of wall and the formation of bubbles ("nucleate boiling"). Due to the flow turbulence of coolant the bubbles will be carried into those more removed from riding-crops and the colder layers where they are condensed. Therefore with nucleate boiling heat fluxes from the wall to the coolant are abstracted/removed more intensively; at the constant rate of coolant the heat-transfer coefficient $\alpha_{\text{о.п}}$

increases/grows two or more times.

However, with further temperature excess $T_{o.п}$ above the boiling point of coolant sharply is increased a quantity of generatrices of bubbles, they not managing to be washed off by coolant flow and to be condensed in its colder layers, but they decant between themselves, forming continuous sheeting pair on the surface of wall ("skin boiling"). In this case the coefficient of heat transfer $\alpha_{o.п}$ and heat flow from the wall to the coolant sharply (10 or more times) they are decreased, which leads to the inadmissible increase in temperatures $T_{в.п}$ and $T_{o.п}$ and to the hot spot of chamber wall.

§11.9. Effect of the type of coolant and parameters of external flowing cooling on the mode/conditions of cooling chamber/camera.

For the effective external flowing cooling of chamber/camera are important to select optimal the type of coolant, its inlet temperature into the coolant passage, and also most advantageous form of the coolant passage, which ensures necessary distribution of the rate of coolant along the length of the coolant passage.

Effect of the type of coolant on the mode/conditions of cooling. The analysis of equation (11.7) shows that heat-transfer coefficient $\alpha_{o.п}$, and, consequently, the cooling ability of different liquids with

one and the same cooling of channel depends substantially on their type. If we assign the rate of coolant, and also the technologically feasible sizes/dimensions and the form of the coolant passage, then for each type of coolant it is possible to determine the value of coefficient $\alpha_{о.п}$, called available; let us designate it $\alpha_{о.п.расп}$.

Page 175.

The necessary value of coefficient $\alpha_{о.п.потр}$ can be determined in equation (11.16), after substituting in it permissible temperatures $T_{о.п}$ and $T_{ох}$.

Normal cooling is provided under the condition

$$\alpha_{о.п.расп} > \alpha_{о.п.потр}.$$

For the evaluation of the cooling properties of coolant have a value the heat capacity of coolant, the temperature range of its liquid state, and also the portion of coolant in the propellant composition, determined by coefficient $\varkappa$.

Temperature range of the liquid state of coolant is determined by a difference in temperatures $T_{кип}-T_{пл}$, moreover boiling point must be taken at the pressure of coolant which it has in the coolant passage. With the increase of the difference in temperatures and specific heat of coolant indicated its cooling capacity is increased.

Under conditions of rocket vehicle for the external flowing cooling of chamber/camera it is possible to use only propellant components. The propellant component flow (and, consequently, of coolant) is limited, but furthermore, not all components possess the sufficiently good cooling properties.

Of all liquids the best cooling capacity possesses the water. The heat-transfer coefficient $\alpha_{\text{о.п}}$ of nitrogen tetroxide and nitric acid is 1.5-2.0 times less, and kerosene and UDMH - are three times less than the value $\alpha_{\text{о.п}}$ of water. Usually as the coolant of chamber/camera of ZhRD serves fuel (kerosene, ammonia, UDMH, hydrogen, etc.), but if it cannot ensure the required cooling, is used oxidizer (for example, nitric acid, nitrogen tetroxide and peroxide of hydrogen).

The advantage of the use/application of an oxidizer as the coolant lies in the fact that its expenditure/consumption 2-4 times usually exceeds fuel consumption, i.e., $\varkappa$=2-4. However, if coolant is oxidizer, then material of internal wall must possess resistance in the oxidative medium at elevated temperatures.

The influence of the temperature of coolant to the

mode/conditions of cooling. The analysis of equation (11.17) shows that with the decrease of the temperature of coolant the temperature of the heated surface of internal wall $T_{\text{в.п}}$ also is decreased, which as was shown in §11.8, it is desirable, in spite of certain increase of total heat flux q_{Σ}.

The temperature of coolant can be lowered, if to decrease value q_{Σ} (see §11.11). It is possible to supercool it to the temperature lower than ambient temperature, with the aid of the special system, which forms part of starting/launching device.

Page 176.

Effect of the rate of coolant on the mode/conditions of cooling. The rate of coolant $W_{\text{ох}}$ in significant degree determines heat-transfer coefficient $\alpha_{\text{о.ж}}$ (see equation (11.7)and, consequently, also heat transfer from the cooled surface of wall to the coolant. In accordance with equation (11.3) with an increase in coefficient $\alpha_{\text{о.ж}}$ increases/grows value q_{Σ}, but temperatures $T_{\text{о.ж}}$ and $T_{\text{в.п}}$ descend (see §11.8).

The rate of coolant in the coolant passage of chamber/camera can be raised, after increasing its flow rate per second $\dot{m}_{\text{ох}}$ when $f_{\text{о.т}}=\text{const}$ or decreasing the flow passage cross-sectional area of

coolant passage $f_{o.T}$ when $\dot{m}_{ox}$=const. Area $f_{o.T}$ they decrease by the selection of the corresponding sizes/dimensions and form of the coolant passage (see §11.11).

With the increase of the velocity of coolant is increased the hydraulic resistance of cooling channel $\Delta p_{o.T}$, what is undesirable. Usually value $\Delta p_{o.T}$ is 5-20 bars [≈5-20 kgf/cm²]. Therefore it is important to select the optimal speed of coolant W_{ox} in different cross sections of the coolant passage. Heat fluxes have greatest value in the critical cross section, and speed W_{ox} in it must be greatest; it can reach 50-60 m/s.

Effect of the area of the cooled surface on the mode/conditions of cooling. If we disregard/neglect the thickness of internal wall δ_{cT}, then in the simplest form of the coolant passage (in the form of the annular slot between the external and internal chamber walls) the area of the heated surface of internal wall $F_{н.п}$ is equal to the area of its cooled surface $F_{o.п}$, i.e. $F_{н.п}=F_{o.п}$.

Cooling efficiency can be raised under condition $F_{o.п}>F_{н.п}$, which is provided in the presence of the edges/fins of one or the other construction/design on the cooled surface of internal wall [7].

During steady state of cooling the value of heat flux, equal to

sum $\dot{Q}_{кон.н}+Q_{л}$, it is time-constant. Therefore when $F_{о.п}>F_{н.п}$ occurs insquality $q_{о.п}<q_{н.п}$, moreover $q_{н.п}=q_{\Sigma}=q_{кон.н}+q_{л}$.

The decrease of value $q_{н.п}$ in comparison with value $q_{о.п}$ is determined on the relationship/ratio

$$\frac{q_{н.п}}{q_{о.п}}=\frac{F_{о.п}}{F_{н.п}}.$$

Using ribbing on the cooled surface of internal wall, it is possible to increase area $F_{о.п}$ 1.4-1.8 times and more in comparison with area $F_{н.п}$; in so many once is decreased the necessary value of heat-transfer coefficient $\alpha_{о.п}$ in comparison with value $\alpha_{о.п}$ for the coolant passage of the simplest form (without the ribbing).

Page 177.

§11.10. Calculation of preheating coolant in that cooling to the channel of chamber/camera coolant continuously receives heat fluxes, its that that temperature along the length of the coolant passage continuously increases/grows and it reaches the greatest value before the entrance in the head of chamber/camera. Depending on the sizes/dimensions of chamber/camera and heating power of fuel/propellant cooling temperature in that cooling of channel is raised on 100-300 deg.

Preheating coolant in each section of the coolant passage is

determined according to the equation

$$T_{ox.вых}=T_{ox.вх}+\Delta T_{ox}. \qquad (11.20)$$

Value ΔT_{ox} is designed as follows.

The quantity of heat, received by coolant in the i section of chamber/camera, is

$$Q_i=q_{\Sigma i}F_i,$$

where $q_{\Sigma i}$ - total specific heat flow in the i section, determined from graph $q_\Sigma=f(l)$ (see Fig. 11.1), which must be preliminary constructed according to the results of calculating the heat fluxes into the wall; F_i - surface of the wall of the i section, through which heat flux is transmitted to coolant.

If in the i section of chamber/camera the temperature of coolant c_{ox} with heat capacity $\dot{m}_{ox}$ is increased on $\Delta T_{ox\,i}$, then the heat flux, received by coolant in this section, can be written in the following form:

$$Q_i=\dot{m}_{ox}c_{ox}\Delta T_{ox\,i}. \qquad (11.21)$$

Consequently,

$$\Delta T_{ox\,i}=\frac{Q_i}{\dot{m}_{ox}c_{ox}}. \qquad (11.22)$$

The heat capacity of coolant c_{ox} depends on its temperature, which changes along the length of the coolant passage. Therefore preheating for each section is designed at mean temperature of coolant by successive approximations, moreover in the first

approximation, they accept, that the temperature of coolant all over length of this section is constant and equal to its temperature at the entry into this section.

The temperature of coolant at the output/yield from the coolant passage is equal to

$$T_{\text{ох.вых}}=T_{\text{ох.вх}}+\frac{\sum_{i=1}^{i=n} Q_i/c_{\text{ох}\,i}}{\dot{m}_{\text{ох}}}.$$

Temperature $T_{\text{ох.вых}}$ in the majority of the cases must not exceed the boiling point of coolant, moreover the latter, as has already been indicated, must be taken at that pressure which a coolant has at the output/yield from the cooling loop.

Page 178.

On the basis of equation (11.21) the maximum heat-absorbing ability of coolant is equal to

$$Q_{max} = \dot{m}_{ox} c_{ox} (T_{ox.вых} - T_{ox.нач}). \qquad (11.23)$$

Examination of equation (11.23) makes it possible to come to light/detect/expose the following ways of an increase in the heat-absorbing ability of the coolant:

a) a reduction/descent in temperature $T_{ox.нач}$, i.e. the utilization of a coolant in the supercooled state;

b) use/application of both propellant components as the coolant.

In a number of cases with the insufficient heat-absorbing ability of coolant they search for ways of reducing/descending the heat fluxes into the chamber walls, i.e., decrease value $\sum_{i=1}^{i=n} Q_i/c_{ox\,i}$.

§ 11.11. The design features of the cooling systems of chamber/camera.

In the preceding paragraphs was examined in essence external flowing cooling. With this method heat fluxes are abstracted/removed from the chamber wall with the aid of the coolant, which takes place through the coolant passage of one or the other form. After the coolant passage the coolant (propellant component) is introduced through the head inside the combustion chamber.

External flow-through cooling is called also regenerative, since in practice entire/all heat, which entered into the internal wall and given up by it to coolant, returns to the combustion chamber and effectively is used (it is regenerated). Furthermore, the preheating of component of fuel/propellant contributes to its more rapid vaporization and more complete combustion in the chamber/camera.

external flowing cooling comparatively rarely is used in the pure form. Usually chamber/camera as a whole or at least its any section is additionally cooled by another method. This cooling is called combined (mixed).

As the example it is possible to give cooling basic part of the chamber/camera by external flowing cooling, and the final part of the nozzle - by radiation/emission.

Construction/design of the coolant passages of chamber/camera.

The effectiveness of external flowing cooling significantly depends on sizes/dimensions and form of the coolant passage which must provide the desired values of the rate of coolant and heat-transfer coefficient $\alpha_{o.n}$ along the length of channel.

Page 17.

Are distinguished two types of the coolant passages:

a) the smooth ring coolant passage, in which outer and internal chamber walls are not connected between themselves along the length of the chamber/camera:

b) the coolant passage with ribbing, in which external and internal chamber walls are fastened between themselves by the edges/fins of one or the other construction/design all over length of chamber/camera.

The smooth ring coolant passage (Fig. 11.4) has simple construction/design and possesses low hydraulic resistance. This channel can be used for low pressure and it is sufficient high expenditure/consumption of coolant.

Are very effective the coolant passages with the ribbing. To the channels with the ribbing and the axial motion of coolant they relate:

a) channel with the longitudinal edges/fins;

b) channel with the adapter, which has longitudinal corrugations;

c) the channel, made from the longitudinal tubes, soldered between themselves on lateral surfaces.

Channels with ribbing and motion of coolant along the helix are the channel with the spiral edges/fins and the channel, formed by helical tubes.

If internal and external walls are connected, then external wall to the certain degree is unloaded. Chambers/cameras with this coolant passage possess high strength and rigidity, which makes it possible to use the walls of small thickness at a sufficiently large pressure in the coolant passage. In the coolant passages with the ribbing to more easily ensure high rate of coolant, than in the smooth ring

channels.

In § 11.9 it was shown that the presence of edges/fins increases heat-transfer coefficient $\alpha_{o.n.}$ Channels (longitudinal or spiral) more evenly distribute coolant over the cross section of channel. Therefore the coolant passages with the ribbing are used extensively in the chambers/cameras ZhRD [- liquid propellant rocket engine] in spite of the complication of their construction/design.

Channel with the longitudinal edges/fins (Fig. 11.5a) fulfill by milling longitudinal edges/fins on the external surface of internal chamber wall and by subsequent connection of edges/fins on the upper ends/faces with the external wall with the aid of the seam welding or rations.

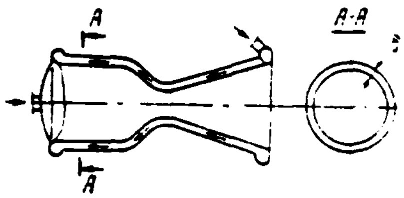


Fig. 11.4. Chamber/camera with the smooth ring coolant passage (δ - height/altitude of the channel of the coolant passage).

Page 180.

Channel with the corrugated adapter (see Fig. 11.5b) is external and internal walls, into radial clearance between which is inserted the adapter with the longitudinal corrugations. The apexes/vertexes of corrugations are connected with the walls by soldering. The corrugated adapter makes it possible to divide coolant flow into two flows and to achieve in this case an increase in the rate of coolant. Furthermore, in this case the collector/receptacle of coolant (fuel) place not at the end the nozzles, but approximately/exemplarily in the middle of expanding section of nozzle, which decreases the length of the conduit/manifold, which supplies fuel to the chamber/camera. In this coolant passage the low part of the coolant flow (20-30o/o) goes along the channels, formed by adapter and external chamber wall, before exit section, and then along the channels, formed by adapter

and internal wall, towards critical cross section. Basic part of the flow goes immediately towards critical cross section throughout the channels, formed by adapter and external wall. In the special collector/receptacle at the entry into critical cross section both flows are mixed and then evenly they enter the channels between the adapter and the external wall, and also between the adapter and the internal wall.

The channel, made from the longitudinal tubes (Fig. 11.6), is the variety of the coolant passage with the ribbing. The most widely used form of the cross section of tubes is rectangular or trapezoidal with the filleted corners. Tubes bent on the duct/contour of chamber/camera. Width and cross-sectional area along the length of tube different.

The ends/faces of tubes solder in the collectors/receptacles for supplying and branch/removal of coolant. One of the advantages of can-type chamber is the possibility to introduce coolant into the channel and to derive/conclude coolant from it from one and its the same end. In this chamber/camera that supplying and offtake collectors/receptacles place at the head of chamber/camera.

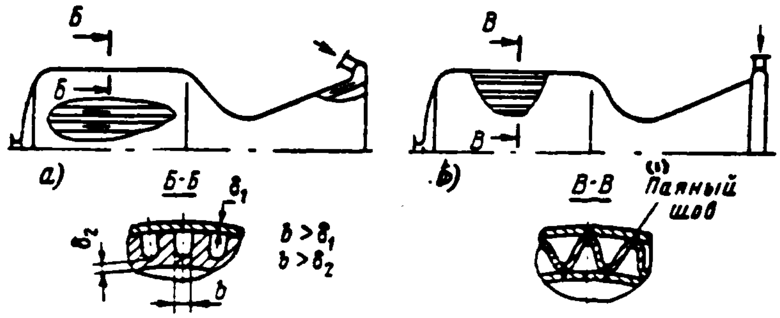


Fig. 11.5. Chambers/cameras with the milled longitudinal channels (a) and the corrugated adapter (b).

Key: (1). Soldered seam.

Page 181.

Coolant makes two courses: in each adjacent pair of tubes the coolant on one tube goes from the head to the nozzle, and on the adjacent - in the opposite direction.

The longitudinal webs of tubes connect between themselves by soldering, so that tubes form chamber wall.

In order to raise the strength of tubular chambers/cameras over their length they place several bands/shrouds/tires (power rings) or is wound chamber/camera by tape or wire from steel or high-strength

alloys, and also by fiberglass.

Can-type chambers possess high strength and rigidity with the relatively low mass; as a result of the effect of ribbing and small wall thickness they reliably are cooled. If in the chambers/cameras with the edges/fins or the corrugated adapters solder can flow in into the channels and cover them, then in the can-type chambers this shortcoming will be eliminated due to the arrangement/position of soldering seams out of coolant channels.

Channel with spiral channels (Fig. 11.7) is used when channels with the longitudinal channels do not assure the required heat-transfer coefficient $\alpha_{o.п}$. Spiral channel can be mono- or multiple. The effect of the use/application of spiral channels lies in the fact that at one and the same values of the height/altitude of channel and expenditure/consumption of coolant its rate more than the rate in the longitudinal channel, this difference increasing/growing with the decrease of a number of approaches. Furthermore, the surface of edges/fins in the channel with the spiral channels is also more than in the channel with the longitudinal channels, which additionally increases cooling efficiency.

However, for the coolant passage with the spiral channels characteristically high hydraulic resistance, and execution of channels, especially in the sections of chamber/camera with the variable/alternating cross section, is complicated.

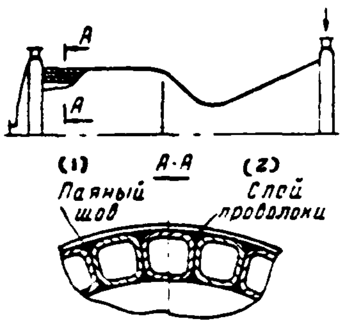


Fig. 11.6.

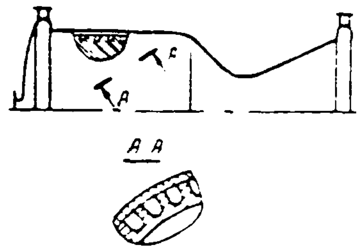


Fig. 11.7.

Fig. 11.6. Chamber/camera, soldered from shaped longitudinal tubes and wrapped by layer of wire.

Key: (1). Soldered seam. (2). Layer of wire.

Fig. 11.7. Chamber/camera with coolant passage, which has helical channels.

Page 182.

Therefore the coolant passages of this construction/design use only in the most heat-stressed sections chambers/cameras, in the first place, in the area of critical cross section.

Chambers/cameras with the the spiral of winding of tubes did not

win acceptance as a result of the significant hydraulic resistance and difficulty of the safeguard of a smooth duct/contour of internal surface (on the generatrix of chamber/camera).

Methods of reducing/descending the heat fluxes into a chamber wall.

Heat fluxes from the combustion products into the wall of chamber can be decreased by the utilization of internal cooling or layer of the thermo-insulating material, applied to the internal surface of chamber wall.

Internal cooling. The cooling, during which the coolant is introduced inside the chamber/camera and is created a wall layer of gas of a reduced temperature, is called internal or film.

For the required reduction/descent in the temperature of a wall layer the expenditure/consumption the supplied combustible is less necessary expenditure/consumption of oxidizer. This can be explained by the greater slope of curve of dependence $T_{газ}=f(\alpha_{ок})$ in region $\alpha_{ок}<1$, than in region $\alpha_{ок}>1$ (Fig. 11.8). Furthermore, working conditions of the heated surface of chamber wall in in recovery/reduction medium are easier than in the oxidative.

The coolant, utilized for the internal cooling, must possess

large heat capacity in the liquid and gaseous state, and also high values of the boiling point, heat of vaporization and dissociation.

The effectiveness of the internal cooling increases, if during the decomposition/expansion of coolant are formed only gaseous products with the low molecular weight. This requirement to a considerable degree satisfies the series/number of the fuels: H_2, NH_3, MMG, etc. The chemical energy of propellant component, which is located in the excess in near-wall layer, is used not completely. Therefore internal cooling to a certain degree decreases the specific impulse of chamber/camera.

The coolant, used for the internal cooling (combustible), is derived/concluded to the heated surface of chamber wall by the following methods:

a) through additional fuel nozzles, placed on the periphery of the head of chamber/camera; b) through the belts/zones of curtain and c) through the belts/zones of porous inserts.

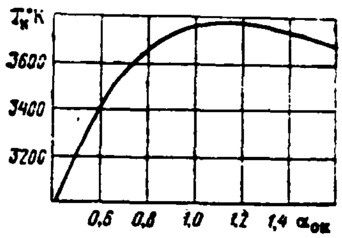


Fig. 11.8. Dependence of the temperature of products of the combustion of fuel O_2+ kerosene on coefficient α_{ok}

Page 183.

The first method is structurally/constructionally most simple: it are been commonly used in the combination with the second method (with the belts/zones of curtain). This is explained by the fact that a wall layer with the excess of the introduced coolant in proportion to motion from the head to the nozzle is mixed with the combustion products.

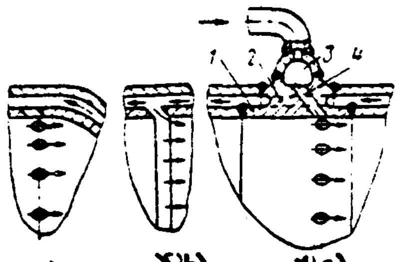
Usually a quantity of belts/zones of curtain does not exceed three, moreover them establish/install before the heat-stressed sections chambers/cameras, in the first place, in the nozzle entry and before the critical cross section.

The belts/zones of curtain are rd the fine/small and of greater partly tangential (tangentially to the cylinder of chamber/camera)

openings/apertures, located in the circumference in this cross section of chamber/camera, or annular slot (Fig. 11.9).

Coolant is supplied into the openings/apertures of belts/zones directly from the coolant passage or from the collector/receptacle to which it is supplied by the special conduits/manifolds (see Fig. 11.9). In the latter case in the ring of curtain are fulfilled two groups of the openings/apertures, displaced in circumference one relative to another. Through the radial (or tangential) openings/apertures, which are the openings/apertures of the belt/zone of curtain, coolant is introduced inside the chamber/camera. The axial openings/apertures (in Fig. 11.9c dotted line showed one such opening/aperture) provide duct coolant through the coolant passage through the ring curtains.

The chambers/cameras of ZhRD of the low thrust (50-5000 n [~~5-500 kgf]), including with multiplying, can be cooled only by internal cooling (without the external flowing cooling). Its effectiveness depends on the properties of components of propellant (in particular the component, utilized as the coolant), and also on the heat resistance of material of chamber wall. The lower the temperature of combustion products, the more effective the coolant and the the large temperature of heating wall allows/assumes its material, the less the necessary expenditure/consumption of coolant and the connected with it losses of specific impulse.



[illegible]. The diagrams of the input/introduction of coolant inside the chamber/camera for organizing the internal cooling: a) through the belt/zone of openings/apertures into the internal wall; b) through the slot belt/zone of curtain; c) through opening into the ring of curtain; 1 - ring of curtain; 2 - oblong hole; 3 - collector/receptacle of curtain; 4 - opening/aperture of curtain.

Page 184.

The final part of the nozzle of the chamber/camera of some ZhRD (for example, JRD-F-1) is cooled by the working medium/propellant of turbine which is introduced inside the nozzle through the collector/receptacle, distant behind nozzle exit section at a distance, which ensures the excess of the pressure of the working medium/propellant above the pressure of combustion products in this cross section of nozzle. Gas from the collector/receptacle is supplied into the nozzle through several slot belts/zones of curtain or belts/zones with the tangential openings/apertures (tangential

input/introduction of gas raises cooling efficiency).

During the cooling with the curtain the final part of the nozzle can be prepared from the usual stainless steel, including for ZhRD with multiplying and significant total operating time. Certain shortcoming in this cooling is the need for pressure increase at the turbine exhaust, which decreases the developed with it power (see § 13.13).

Coolant can be supplied inside chamber through the wall from the porous material. In this case the coolant under the pressure continuously acts on the numerous smallest pores which evenly distributed by entire volume of wall, and creates on the heated surface of wall a layer of the liquid or evaporated coolant. This cooling is called porous.

The difficulties of designing of chamber/camera with the porous cooling are explained by the complexity of obtaining uniform porosity of wall, by the low strength of porous materials and by the possibility of soiling pores during the work of engine. Therefore this cooling it is expedient to use only for the chambers/cameras with the increased calorific intensity.

coating of thermo-insulating material on the internal surface of

chamber/camera. The effect of the use/application of a layer of thermo-insulating material in addition to external flowing cooling consists of the following. In the case of the high melting point of thermo-insulating materials it is possible to allow of high heating the surface of its layer, washed by combustion products, which decreases the heat fluxes into the wall and preheating of coolant in the coolant passage. Furthermore, due to the low coefficient of thermal conductivity the temperature of a layer of thermo-insulating material sharply falls according to its thickness. Therefore the temperature of the surface of wall, to which will be brought in the layer indicated, noticeably lower than temperature of chamber/camera without the thermal insulation (Fig. 11.10).

As thermo-insulating materials can serve oxides of refractory metals (dioxide of zirconium ZrO_2, oxide of magnesium MgO, oxide of aluminum Al_2O_3) and their carbides, molybdenum disilicide $MoSi_2$ and so forth.

The thickness of the layer of the materials indicated which most frequently will be brought in by plasma spraying, is 0.3-0.6 mm. For the best adhesion (cohesion/coupling) of a layer with the surface of chamber wall to it preliminarily will be brought in the substratum of chromium or nickel with thickness of up to 0.1 mm.

Are most finished thermo-insulating coatings from the dioxide of zirconium and molybdenum disilicide.

A layer of thermal insulation works under severe conditions.

Page 185.

Therefore the creation of chamber/camera with a layer of thermal insulation causes large difficulties; in the layer indicated frequently are formed the cracks and crumbling off in the series/number of sections. Coating of thermal insulation on the internal surface of chamber/camera complicates its manufacture and increases its cost/value and mass.

Other methods of wall cooling.

Let us examine ablation and radiant cooling of the finite segment of nozzle or entire chamber/camera.

Ablation cooling. Ablation cooling is called the cooling, provided by a layer of material which will be brought in to the internal surface of chamber/camera and in the process of chamber operation undergoes so called ablation. Ablation is the involved complex of the processes, which take place with the ingress of heat

and leading to the destruction of surface layer. Such processes are the processes with the phase transformations (melting, vaporization, sublimation) and the processes of decomposition; the heat, spent on their course, is called the heat of ablation. As a result of ablation are formed the gaseous and solid products, which create a wall layer with a reduced temperature and are taken away by the flow of products combustion. Therefore thickness of the layer of material, applied to the wall, in the process of chamber operation continuously is decreased.

The material, which undergoes ablation, is called ablating (or destroying). Ablation cooling is called also cooling via the ablation of substance.

The heat fluxes, entering into layer of the ablating material, go in essence to the maintenance of ablation, so that the heat flux, passing through a layer of the ablating material, is not great.

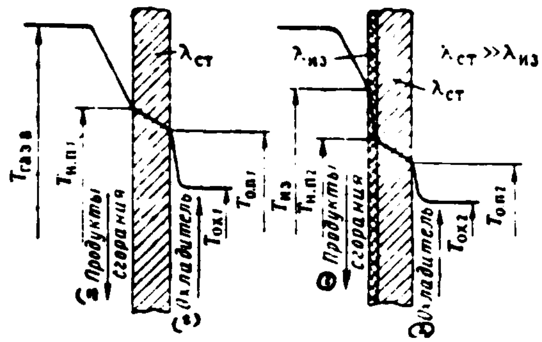


Fig. 11.10. Graphs of the temperature distribution on thickness of chamber wall with a layer of thermal insulation and without it.

Key: (1). Products of combustion. (2). Coolant.

Page 186.

On the surface of the layer indicated is established a comparatively low temperature (several hundred degrees), which depends on the composition of the ablating material.

As the ablating material can serve fibers or fabrics from oxide of silicon, graphite, carbon, asbestos and quartz, impregnated with phenolic resin. Chambers/cameras with the ablation cooling have a series/number of advantages before the chambers/cameras with the external flowing cooling. Such advantages they are:

a) the absence of the coolant passage, which simplifies chamber design, it decreases hydraulic losses in main of one of the propellant components and decreases the possibility of its freezing under conditions of outer space;

b) the admissibility of a substantially great [illegible] coefficient α of the temperature of the components of propellant and pressure of combustion products (and, consequently, of the thrust of chamber/camera) under the condition for reliable cooling.

However, to chambers/cameras with the ablation cooling are specific essential shortcomings, namely:

a) the limitation of the value of specific impulse; with its increase must be increased the thickness (and, consequently, mass) of a layer of the ablating material;

b) the limitation of the operating time of engine; for the prolonged work of engine is required the large thickness of the layer of the ablating material;

c) the need for the account of an increase in the cross-sectional area of the nozzle (especially critical cross section), called by the decrease of the thickness of the layer of the

ablating material.

Ablation cooling is used in essence for the chambers/cameras with the small values of thrust and pressure p.

Radiant cooling. During the radiant (or radiation) cooling the heat from the chamber walls is abstracted/removed by radiation/emission into the surrounding space. The heat fluxes, passing through the wall of this chamber/camera and radiated into the surrounding space, are comparatively low. Therefore in accordance with equation (11.14) wall has sufficiently high temperature (to 1800°K and more).

For the chambers/cameras with the radiant cooling is characteristic prolonged (to 60 s and more) operating cycle in the nonstationary system of cooling. At the end of the period indicated is established the equilibrium temperature of wall, since begins the equality the heat fluxes, which come the wall and abstracted/removed from it.

The utilization of radiant cooling in a number of cases makes it possible to significantly decrease the mass of chamber/camera (among other things in comparison with the chamber/camera, which has ablation cooling), especially with the long operating time of engine.

Shortcomings in the radiant cooling are reed the uses/applications of the expensive high-temperature (strength) alloys, the manufacture of parts from which is complicated.

Page 137.

Furthermore, these alloys are brittle and possess low chemical stability to the combustion products. In order to avoid the oxidation of such alloys by combustion products, to the internal chamber wall will be brought in special coating; for example, wall from the niobium alloy they cover/coat with a layer of organosilicon compounds.

In a number of cases the coating not only shields the surface of wall from the oxidation, but also increases its radiating capacity, which makes it possible to additionally lower the temperature of wall. Such properties possesses, in particular, the oxide film of aluminum, applied to the surface of wall from the nickel alloy.

Uncooled chambers with the massive wall. Normal conditions for chamber operation can be ensured, using heat capacity of material of its wall. If chamber wall possesses large mass, and its material by

large heat capacity and by thermal conductivity, then wall can receive the heat fluxes which are distributed all over mass until the temperature of wall achieves the maximum permissible for this material value. Such chambers/cameras (them call also uncooled or cooled with the aid of "sponge" coolings) they use in essence in bench experimental of ZhRD.

Chapter XII.

CHAMBERS/CAMERAS OF LIQUID PROPELLANT ROCKET ENGINES.

§ 12.1. General/common/total characteristic of chambers/cameras.

Chamber/camera of ZhRD is its basic and most heat-stressed aggregate which to a considerable extent determines perfection and reliability of engine and DU as a whole.

Chamber/camera ZhRD, which works on the diagram "liquid - liquid", consists of head, combustion chamber and nozzle.

Head must introduce propellant components inside the chamber/camera in such a way that the chemical reactions of their reaction would occur fully and into the short time interval.

In the combustion chamber (decomposition/expansion) occurs the vaporization, the mixing of propellant components and their combustion (decomposition/expansion). The combustion chamber volume must be as far as possible low, but sufficient for the safeguard of complete combustion of propellant components prior to the nozzle entry. Volume of combustion chamber count the volume of

chamber/camera from the internal (fire) bottom heads to the critical cross section. The length of combustion chamber also affects the completeness of the burning of propellant components, but to a lesser degree than volume.

Nozzle accelerates/disperses combustion products to the greater possible rate for obtaining the high specific impulse of chamber/camera.

Page 188.

The most widely used type of the chamber/camera of two-component ZhRD, which work on the diagram "liquid - liquid", is cylindrical chamber/camera with the coolant passage and the head, which has three bottoms (Fig. 12.1). Oxidizer is supplied through intake pipe 1 into the cavity and between external bottom 2 and average/mean bottom 3, and from it through injectors 11 - inside the combustion chamber.

Fuel is supplied through intake pipes 7 (their -usually two) into collector/receptacle 8, usually arranged/located on certain distance from nozzle exit section (see § 11.11). Spreading on the collector/receptacle, fuel enters the coolant passage c, formed by external wall 5 and internal wall of 6 chambers/cameras. The flow of fuel is divided into two parts: basic part heads toward the head of

chamber/camera, and other - to rotary collector/receptacle 9 at the end of the nozzle and after rotation along the corresponding channels also to the head. From the coolant passage the fuel enters cavity b between average/mean bottom 3 and fire bottom 4, while from it through injectors 10 - inside the combustion chamber.

Chamber/camera ZhRD, which work on the diagram " gas- liquid" and "gas- gas", consists of head, afterburner (in certain cases - combustion chambers) and nozzle.

As it was shown into § 9.1, in the diagram "gas - liquid" into the chamber/camera of precombustion are supplied generator gas and liquid propellant component, and in the diagram "gas- gas" - reducing and oxidative gases of two ZhGG.

During the design and the construction the chambers/cameras must first of all be provided;

a) high reliability;

b) large specific impulse;

c) low mass with the sufficient strength;

d) small sizes/dimensions, especially along the length, since the length of chamber/camera determines the length of engine as a whole.

Chambers/cameras of ZhRD differ from each other in terms of the combustion chamber configuration, in terms of the type of head and used in it injectors, in terms of the type of the nozzle (see Chapter VI), in terms of the method of the cooling (see Chapter XI) and in terms of other special features/peculiarities.

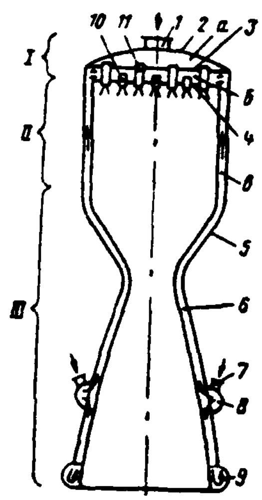


Fig. 12.1. Diagram of cylindrical chamber/camera with coolant passage: I - head: II - combustion chamber; III - nozzle; 1 - intake pipe of oxidizer: 2 - external bottom of head: 3 - average/mean bottom of head; 4 - internal (fire) bottom; 5 - outer wall; 6 - internal wall; 7 - intake pipe of fuel; 8 - inlet manifold of fuel; 9 - rotary collector/receptacle of fuel; 10 - fuel nozzle: 11 -

oxidizer nozzle.

Page 189.

§ 12.2. Combustion chamber configurations (afterburnings).

On the geometric combustion chamber configuration (afterburning) they subdivide into the cylindrical ones, shaped, spherical ones and ring ones (Fig. 12.2).

Most extensively use for the engines of the most varied thrusts cylindrical combustion chambers (Fig. 12.3). They are simple by the construction/design and they are not complex in the manufacture. The constancy of cross-sectional area along the length of such chambers/cameras makes it possible to organize the effective combustion of propellant components; in particular, is eliminated the formation of the stagnation zones, in which does not proceed the process of burning. The relatively low outside diameter of cylindrical combustion chambers lightens their use/application in multichamber of ZhRD or in the engine installation, which consists of several single-chamber engines.

To shortcomings in the cylindrical combustion chambers in the comparison with the spherical ones they relate:

a) the lowered/reduced strength characteristics, which forces to increase wall thickness;

b) the greater hydraulic resistance of the coolant passage;

c) the increased surface of walls which must be cooled.

Distinguish isobaric and high-speed/high-velocity chambers/cameras. Isobaric are called combustion chambers, in which the pressure of combustion products over their length remains approximately/exemplarily constant; the ratio of cross-sectional area to the throat area of such chambers/cameras $f_к/f_{кр}>3$.

The relation indicated call relative area of chamber of combustion and designate $\bar{f}_к$, i.e.

$$\bar{f}_к=\frac{f_к}{f_{кр}}. \qquad (12.1)$$

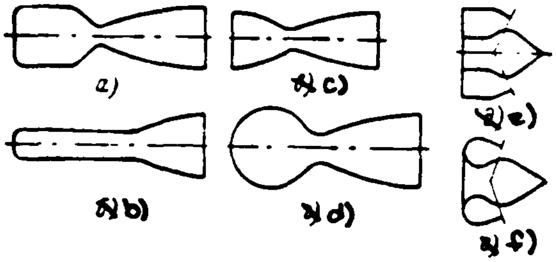


Fig. 12.2. The combustion chamber configurations: a) are cylindrical; b) semi-heat nozzle; c) in the form of the shaped tapering portion; d) spherical; e) ring cylindrical with the inner body; f) ring toroidal with the inner body.

Page 190.

Chambers/cameras with value $f_{к}<3$ call high-speed/high-velocity. They possess the so-called thermal resistance: the stagnation pressure of gas at the end of such combustion chambers is less than at their beginning; this effect is caused by the delivery of heat to the flow of gas, which moves in cylindrical pipe [28]. With the decrease of value $f_{к}$ the gas velocity and the thermal resistance of chamber/camera are increased, leading to the appropriate decrease of its specific impulse. Furthermore, with an increase in the velocity of combustion products increase/grow losses of pressure due to friction of motion in the combustion chamber. Therefore for safeguard of one and the same pressure of combustion products at the nozzle

entry with the decrease of value f_{κ} must respectively raise the pressure of components of the propellant, with which they are supplied into the combustion chamber.

Limited application found cylindrical combustion chambers, in which value f_{κ} is equal to one (see Fig. 12.2b); then they call semi-heat nozzle.

In proportion to perfection ZhRD is increased pressure p_{κ}, they are improved cooling chamber/camera and the construction/design of head simultaneously is decreased its outside diameter, are used new components of propellant and structural materials. In this case is decreased the combustion chamber volume and increase/grow the sizes/dimensions of nozzle.

In certain cases is used the shaped inswept combustion chamber, in which simultaneously occurs the combustion of propellant components and the dispersal/acceleration of combustion products to the critical speed (see Fig. 12.2c).

Spherical combustion chambers (see Fig. 12,2d) possess the smallest surface with the prescribed/assigned volume, which lightens cooling chamber/camera and it makes it possible to decrease its mass, including as a result of smaller necessary thickness of walls.

However, in such combustion chambers it is most difficult to ensure the even distribution of the expenditure/consumption of combustion products according to the cross section, and in the zone of head can be formed stagnation zones.

Together with the spherical ones can be used the combustion chambers of pear-shaped and elliptic form.

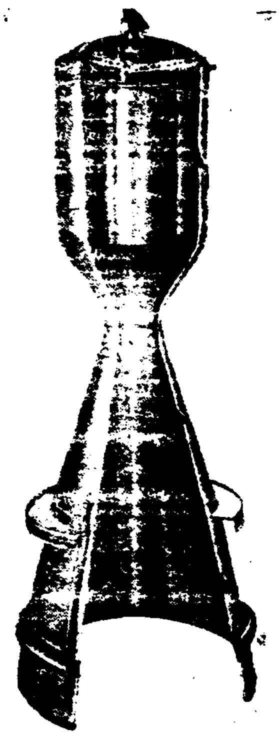


Fig. 12.3. Chamber/camera ZhRD RD-107 the "East".

Page 191.

The injectors of such chambers/cameras place on the flat/plane bottom, and also on the precombustion chambers (antechambers), which make it possible to increase surface for positioning/arranging the injectors.

In connection with the relative complexity of

construction/design and technology of the manufacture of spherical combustion chambers and the absence of their perceptible advantages before the cylindrical ones spherical combustion chambers found limited application in ZhRD.

Annular combustion chambers have a form of the cylindrical ring (see Fig. 12.2e) or the torus/Torr (see Fig. 12.2f).

Annular combustion chambers together with the nozzle of external expansion (or by nozzle with the inner body) possess the series/number of essential advantages in comparison with the usual chambers/cameras. Bases of them are examined in chapter 6. Other advantages include convenience in the arrangement/position of the aggregates of the feed system of propellant components within the inner body of chamber/camera and possibility of designing of efforts/forces for the flight control of rocket vehicle (during the sectional construction/design of combustion chamber).

The greatest effectiveness of ZhRD with the annular combustion chambers is assured with their work on the high-energy of fuel/propellant (first of all on fuel/propellant O_2+H_2 or F_2+H_2).

§ 12.3. Injectors.

Liquid propellant components are introduced inside the combustion chamber through the injectors which provide also the atomization of propellant components, which is accompanied by a significant increase in the surface of drops.

Are distinguished two basic types of the injectors: jet and centrifugal.

The jet injectors are the small accurately carried out openings/apertures in the fire bottom of head. Such injectors are prepared also in the form of separate parts with their subsequent soldering into the heads; in this case of injector they differ little from each other.

The jet injectors inject liquid in the form of the parallel or colliding streams (Fig. 12.4).

Output nozzle orifice is called nozzle. The liquid jet, which escape/ensues from the nozzle, is at certain length from it solid cone with small (5-20°) angle at the apex/vertex. Stream decays into the fine/small drops under the effect of the friction of stream against the products of combustion and transverse vibrations, which appear in it.

The major advantage of head with the jet injectors is its relative simplicity and large capacity.

The capacity of head is called the propellant component flow, passing through unity of the surface of its bottom with the prescribed/assigned injector pressure drop.

Page 192.

The jet injector has smaller sizes/dimensions than centrifugal. Therefore per unit of injector face it is possible to place a greater quantity of the jet injectors, than centrifugal ones. Furthermore, the coefficient of the expenditure/consumption of the jet injectors (see pg. 200) 2.5-3 times more than the coefficient of the expenditure/consumption of the swirl injectors. The jet injectors provide the relatively greater range of streams and the smaller thinness of atomization, than centrifugal.

Injectors with the colliding streams (see Fig. 12.4b) give finer/smaller atomization and smaller length of the zone of atomization, than injector with the parallel streams. But the capacity of head with the colliding streams is less than at the head with the parallel streams.

The block of injectors with the colliding streams can consist of two, three, four or five jet injectors, moreover they can be used:

a) the blocks of nozzles of oxidizer;

b) the blocks of fuel nozzles;

c) blocks with the oxidizer nozzles and fuel; the latter in a number of cases they provide the best characteristics in comparison with blocks indicated above.

The block of injectors, in which there are only oxidizer nozzles or only fuel nozzle, is actually the monopropellant injector, while the block of the oxidizer nozzles and fuel - by the duplex-fuel nozzle.

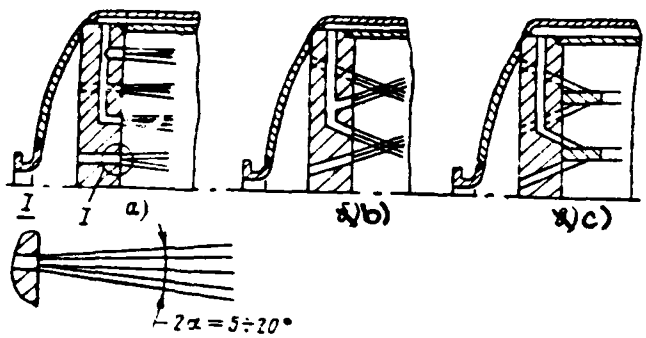


Fig. 12.4. Flat/plane injector assemblies with the jet injectors: a) with the parallel streams; b) with the colliding streams; c) with baffle plates.

Page 193.

The streams of oxidizer and fuel can be supplied to flat/plane baffle plate (see Fig. 12.4c); the thin liquid films, which are generated with the spreading of the streams of propellant components on the plate, collide, providing their good fragmentation and mixing.

The variety of the jet injectors are the slit injectors. Their nozzle has a form of annular slot, but not circle.

In the two-component slit injectors (Fig. 12.5) annular slots are inclined at an angle to the axis of injector, so that liquid jets in the form of two hollow spray cones collide between themselves.

The jet injectors more frequently use for the hypergolic fuels, and also for the chambers/cameras with the low area heads. They are more suitable for the atomization of propellant components with the relatively low viscosity/ductility/toughness.

Centrifugal are called the injectors, in which occurs the torsion of liquid; the liquid jet, which escapes behind their nozzle, is the thin conical film with the angle at the apex/vertex to 120°, which easily decays into the smallest drops.

Centrifugal injectors are subdivided into the tangential ones and the auger ones.

In the tangential injectors (Fig. 12.6b) the liquid is twisted by its input/introduction through one or several tangential openings/apertures, i.e., the openings/apertures whose axis/axle is directed tangentially toward the cylinder of the internal cavity, called the chamber/camera of twisting.

In the auger injectors (or injectors with the swirler) (see Fig. 12.6a) liquid it is twisted due to its motion along the spiral channels, cut on the worm screw (or swirler); liquid enters them from the rear end/face of worm screw.

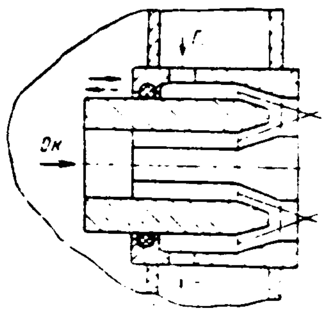


Fig. 12.5. Two-component slit injector with the colliding cones and the variable area of the injection (mechanism for displacing the stock/rod is not shown).

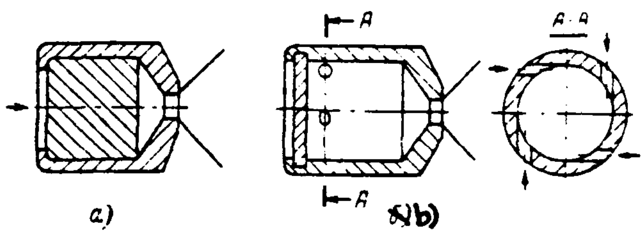


Fig. 12.6. Swirl injectors: a) auger (with swirler): b) tangential.

Page 194.

The swirl injectors provide finer/smaller atomization and smaller length of the zone of atomization, than jet. Their shortcomings are relative structural/design complexity and smaller capacity.

Centrifugal injectors analogous with jet ones are subdivided

into the one-component ones and the two-component ones. In the two-component swirl injectors (Fig. 12.7) the propellant components can be mixed both within the injector (internal mixing) and outside it (external mixing). Injectors with the internal mixing frequently are used for the chambers/cameras, which work on the nonspontaneously combustible fuel/propellant.

In the combined duplex-fuel nozzles are combined jet and swirl injectors; in the injector, depicted in Fig. 12.8, slot fuel nozzle is placed around the auger centrifugal oxidizer nozzle.

The example of the composite injector is also injector with the worm screw, in which there is an axial opening/aperture, be jet injector with small angle of spray cone and large range.

The use/application of the duplex-fuel nozzles decreases the length of the zone of atomization, since the components of fuel/propellant in essence are mixed even in the liquid phase and therefore more rapidly they burn. Furthermore, the capacity of head with the duplex-fuel nozzles is higher than at the head with the one-component swirl injectors.

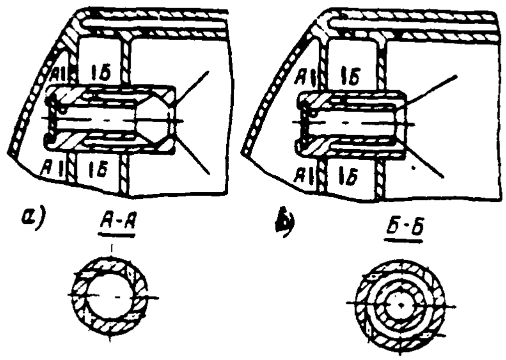


Fig. 12.7

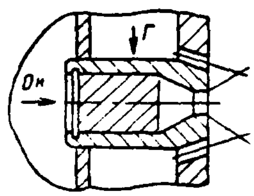


Fig. 12.8.

Fig. 12.7. Two-component swirl injectors: a) with internal mixing; b) with external mixing.

Fig. 12.8. Twc-component ccmpcsite injectcr.

Page 195.

However, the duplex-fuel nczzles are characterized by the increased structural/design ccmplexity; their use/applicaticn leads to the more severe temperature ccnditions for the wcrk of head, as a result of the approximaticn/apprcach to it of flame front due to the decrease of the length of the zcne cf atomization.

The propellant compcnent flcw through the monopropellant

injector is within the limits of 30-300 g/s, and for the duplex-fuel nozzles it can reach 2.5-3 kg/s. Peripheral fuel nozzles usually fulfill with the greater range even smaller to by 20-30c/o expenditure/consumption in comparison with the basic injectors. Expenditure/consumption through the oxidizer nozzles, arranged/located on the periphery of head, is also less than its expenditure/consumption through the basic injectors.

All injectors examined above have the invariable area of nozzle. For the engines whose thrust it is necessary to change over a wide range, use injectors with the variable area the nozzles on which it is possible to withstand/maintain an approximately/exemplarily constant drop/jump in pressures with the considerable decrease of the propellant component flow. The area of nozzle in such injectors can be changed with the displacement/movement of special stock/rod within the injector along its axis/axle and with the overlap of blast nozzle to a certain degree. In the two-component slit injector with the displacement/movement of one stock/rod changes the area of the nozzle of oxidizer and fuel (see Fig. 12.5). Possibly the use/application of other constructions/designs of injectors with their variable area it puffed.

§ 12.4. Heads of chambers/cameras.

The head of chamber/camera serves for the input/introduction and the even distribution of propellant components according to the cross section of combustion chamber.

For the effective vaporization, the mixing and the combustion of the components of propellant and reliable chamber operation its head must provide:

a) the thin and uniform atomization of propellant components, i.e., their fragmentation to the smallest particles, as far as possible differing little from each other in the sizes/dimensions;

b) the identical value of coefficient $\varkappa$ in entire cross section, with exception of a wall layer (Fig. 12.9).

The value of coefficient $\varkappa$ in a wall layer, which corresponds to the excess of fuel, must be also in the possibility of constant on the perimeter of combustion chamber.

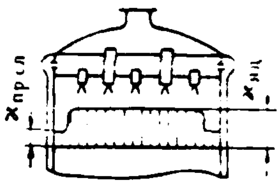


Fig. 12.9. Graph of a change in coefficient $\varkappa$ in the radius of combustion chamber ($\varkappa_{яд}$ and $\varkappa_{пр.с}$ - coefficient $\varkappa$ for the flow core of the products of combustion and wall layer respectively).

Page 196.

For satisfaction of these conditions in the head it is necessary by correspondingly to place the greater possible quantity of injectors.

The important requirement, presented to the head of chamber/camera, is the uniform specific weight flow of propellant components over entire cross section of combustion chamber.

The average/mean over the cross section specific weight flow of combustion chamber is called the ratio of the propellant component flow per second to the area of its cross section

$$r=\frac{\dot{m}}{f_{к}}, \quad kg/(cm^2 \cdot s) \tag{12.2}$$

For the section of cross section with area of Δf_i, the propellant

component flow through which is equal to $\Delta \dot{m}_i$, the local specific weight flow

$$r_i = \frac{\Delta \dot{m}_i}{\Delta f_i}.$$

The hydraulic losses, connected with the delivery of propellant components to the orifices of injector, must be low. Furthermore, head must possess sufficient strength and rigidity, in spite of weakening of its bottoms by a large quantity of openings/apertures under the injectors, and to also provide the smooth starting/launching of the chamber/camera (see § 14.1) and the stable process of burning in it (see § 15.1).

Most extensively are used flat heads (see Fig. 12.3). In them are used the jet injectors with parallel or impinging jets (see Fig. 12.4), and also the swirl injectors (see Fig. 12.7).

Flat heads are simple by the construction/design, are not complex in the manufacture and make it possible to provide uniform specific weight flow over cross section and required distribution of coefficient $\varkappa$ in the radius combustion chambers.

Certain shortcoming in flat heads is their relatively low strength and rigidity. Especially this relates to the chambers/cameras with a large diameter of; therefore between their external and average/mean bottoms vary ring and radial stiffening

ribs, and external base fulfill in the form the parts of the sphere (see Fig. 12.3).

One of the methods of maintaining the necessary conditions for atomization and stable fuel combustion in the combustion chamber with the significant decrease of its flow rate and in the invariable area blast nozzle is the supply of inert gas into the cavity of the head (i.e. it is direct into the propellant components). In this case for the even distribution and the mixing of the liquid components of propellant and bubbles of inert the phase before the injectors are placed special grids.

Page 197.

The construction/design of head in many respects determines reliability and specific impulse of chamber/camera and engine as a whole. During the unsuccessful constructions/designs of heads are noted the following defects of chambers/cameras and the undesirable consequences:

1) erosion or the hot spot of chamber walls, in the first place, in the critical cross section, and also excessive heat fluxes into the walls, about which testify the tracks of the hot spots of wall;

2) the erosion of the internal surface of fire bottom and ends/faces of injectors due to the effect of hot combustion products on them;

3) the unstable process of fuel combustion;

4) the lowered/reduced specific impulse of chamber/camera.

In proportion to the decrease of sizes and thrust of chamber/camera the effect of head on the specific impulse and the stability of the process of burning increases/grows.

In order to lower heat fluxes into the chamber walls it is created, as it was shown into § 11.11, a wall layer of combustion products with a reduced temperature.

For the exception/elimination the erosions of the internal surface of fire bottom and ends/faces of injectors increase a quantity of fuel nozzles in the places for erosion, is used porous material for manufacturing of fire bottom and housing of injectors or they will bring in on them a layer of thermo-insulating material.

The processes of the atomization of the components of propellant, and also their vaporization, mixing and combustion are

not yet studied to such degree that would have the capability to theoretically determine the optimal type of head. Therefore during the development of engine it is necessary to produce the tests of several versions of small scale models and full-scale heads, including firing tests of heads in the composition of chamber/camera.

For the initial tests frequently are taken the heads, which provide only the moderate specific impulse, but they are most reliable. This makes it possible to conduct the tests of engine as a whole in parallel with the finishing of head and chamber/camera. In the course of finishing final selection falls on the head whose construction/design gives the possibility to obtain the greatest specific impulse during the stable fuel combustion.

In a whole series of the cases the necessary combustion stability and the reliable cooling of chamber/camera is achieved only by certain reduction/descent in the specific impulse.

The adjustment of the construction/design of head is the complex and expensive stage of works during the creation of engine.

§ 12.5. Methods of positioning/arranging the injectors on flat heads.

The even distribution of oxidizer and fuel according to the

cross section of combustion chamber is achieved by the appropriate arrangement/position of injectors on the head. There are several methods of positioning/arranging the injectors: checkered, honeycomb, on the concentric circumferences and group.

Page 198.

With the checkered (Fig. 12.10a) injector grid of fuel and oxidizer their quantity is approximately/exemplarily identical: to one fuel nozzle falls one (4x1/4) oxidizer nozzle. Since mass oxidizer consumption usually is 2-4 times more than fuel consumption, then with the staggered arrangement the flow rates through the oxidizer nozzles and fuel considerably are distinguished, which adversely affects carburetion.

During the honeycomb (see Fig. 12.10b) arrangement/position each fuel nozzle is surrounded by several oxidizer nozzles: to one fuel nozzle fall two (6x1/3) oxidizer nozzles. The flow rates through the injectors differ comparatively little, which improves the carburetion of propellant components.

With the injector grid on the concentric circumferences (see Fig. 12.10c) on the head are alternated the circumferences with the fuel nozzles and the oxidizer nozzles. On the circumference, which is

located on the periphery of head, are arranged/located the fuel nozzles, which create a wall layer with lowered/reduced temperature.

During the group arrangement/position the injectors form groups, in each of which is included the specific quantity of oxidizer nozzles and fuel (for example, in ratio 4: 1 or 3: 2) with one and their the same mutual arrangement.

The duplex-fuel nozzles usually place on the concentric circumferences.

The distance between the swirl injectors is determined by the size/dimension of injector itself, and also by the conditions of the strength of head, which is decreased by drilling under the injectors. The distance indicated is selected in the limits of 12-30 mm. The jet injectors place at the substantially smaller distance apart - to 3-4 mm.

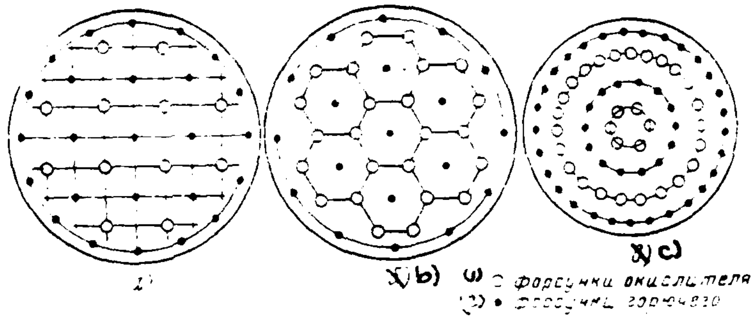


Fig. 12.[illegible]. Types of the arrangement/position of injectors on flat heads: a) checkered; b) honeycomb; c) on the concentric circumferences.

Key: (1). Oxidizer nozzles. (2). Fuel nozzles.

Page 199.

§ 12.6. Calculation of the head of chamber/camera.

For calculating the head must be known following data:

1) density and the viscosity/ductility/toughness of propellant components at a nominal temperature with which they enter injectors;

2) the total oxidizer consumption and fuel;

3) the diameter of the head of chamber/camera; for the cylindrical chamber/camera it is equal to the diameter of combustion chamber;

4) the pressure differential on injector Δp_{Φ}, i.e. pressure difference in the cavity of oxidizer or fuel of head and in the combustion chamber.

An injector pressure drop usually selects in the limits 3-5 bars [~~3-5 kgf/cm²], and in some ZhRD - to 30 bars [~~30 kgf/cm²]. With low pressure differentials deteriorates the atomization of propellant components, and the process of burning becomes unstable. On the other hand, an excessive increase in value Δp_{Φ}, without improving substantially the atomization of propellant components, is caused the need for an increase in the power of feed system.

In ZhRD with the large range of a change in the consumption of fuel m it is necessary to select large injector pressure drops so that and with the work with the low flow rate of $\dot{m}$ (and, consequently, by low value Δp_{Φ}) would be reached the necessary atomization of the stream of fuel/propellant.

A quantity of oxidizer nozzles and fuel which can be placed on the head with its prescribed/assigned diameter, is determined

graphically, after selecting the method of positioning/arranging the injectors and the distance between them (see § 12.5).

Let us introduce the following designations:

$n_{ок}$ and $n_г$ - number of oxidizer nozzles and fuel;

$\dot{m}_{ок.ф}$ and $\dot{m}_{г.ф}$ - per-second flow rate through the oxidizer nozzle and the fuel nozzle.

Values $\dot{m}_{ок.ф}$ and $\dot{m}_{г.ф}$ determine from the formulas

$$\dot{m}_{ок.ф}=\frac{\dot{m}_{ок}}{n_{ок}}; \qquad \dot{m}_{г.ф}=\frac{\dot{m}_г}{n_г},$$

where $\dot{m}_{ок}$ and $\dot{m}_г$ - oxidizer consumption and fuel per second through the head of chamber/camera; they are known from its thermal design.

Page 200.

Calculation of the jet injector.

We use the following known from the hydraulics formulas for the escape of an incompressible fluid from the opening/aperture:

$$W=\sqrt{\frac{2\Delta p_ф}{\varrho}}; \qquad (12.3)$$

$$\dot{m}=\mu W f \varrho, \qquad (12.4)$$

where W - injection velocity of liquid propellant component into the combustion chamber; usually W=15-40 m/s;

$\dot{m}$ - the flow rate per second of liquid propellant component through the head;

f - total area of blast nozzles;

μ - coefficient of flow rate, which considers jet contraction and decrease of real injection velocity in comparison with the theoretical due to the hydraulic resistance.

The coefficient of the flow rate μ of the jet injector depends on the following factors:

a) geometry of the entering edge of opening/aperture; for the sharp edge, especially in the presence of barbs, coefficient μ is less than for the edge with the bevel/facet or the smoothly rounded edge;

b) the purity/finish of machining hole; the large roughness of bore surfaces leads to a considerable reduction/descent in the value μ;

c) the relation of the length of injector l_Φ to diameter of its nozzle d_c, i.e. relation l_Φ/d_c.

In the sharp entering edge and relation $l_\Phi/d_c=0.5\div1.0$ the coefficient of flow rate μ is equal to 0.60-0.65. With an increase in relation l_Φ/d_c to 2-3 value μ increases from 0.6 to 0.75-0.85; simultaneously are increased losses to the friction. It is expedient to select the geometric characteristics of jet injector, with which is provided the greatest coefficient of flow rate. This condition satisfies the injection opening/aperture, shown in Fig. 12.11.

For the determination of the area of the fuel injection or oxidizer let us substitute in equation (12.4) expression W from formula (12.3):

$$\dot{m}=\mu f\sqrt{2\Delta p_\Phi\varrho}, \tag{12.5}$$

whence

$$f=\frac{\dot{m}}{\mu\sqrt{2\Delta p_\Phi\varrho}}. \tag{12.6}$$

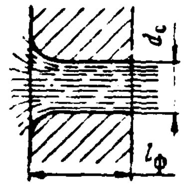


Fig. 12.11.

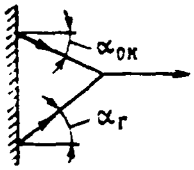


Fig. 12.12.

Fig. 12.11. Injection opening-aperture, which ensures greatest of flow rate (μ=0.85-0.90 when $l_ф/d_c>3$)

Fig. 12.12. Diagram of collision of streams of oxidizer and fuel.

Page 201.

The diameter of blast nozzle usually is selected in limits $d_c=0.5\div3.0$ of mm. The nozzles of smaller diameter it is technologically difficult to perform and, furthermore, they can be obstructed. However, are conducted the investigations of micro-injectors ($d_c<0.25$ мм), of the ensuring better mixing propellant components and large completeness of their combustion. Under condition $d_c>3.0$ мм with more difficulty to obtain the thin atomization of the stream, which escapes behind blast nozzle.

After determining by graphic method examined above a number of

oxidizer nozzles and fuel, it is possible to calculate the area of their openings/apertures (nozzles)

$$f_{ок.с}=\frac{f_{ок}}{n_{ок}};\qquad f_{г.с}=\frac{f_{г}}{n_{г}}.$$

For the head with the colliding streams of oxidizer and fuel angles $\alpha_{ок}$ and $\alpha_{г}$ (Fig. 12.12) select in such a way that that resulting of stream would be parallel to axis/axle chamber/camera. Since the flow rates through the oxidizer nozzles and fuel, and also speed of their injection differ from each other, the condition indicated above is reduced to the equality, which escape/ensues from the law of conservation of momentum

$$\dot{m}_{ок.ф}W_{ок}\sin\alpha_{ок}=\dot{m}_{г.ф}W_{г}\sin\alpha_{г}. \qquad (12.7)$$

By one of the angles they are assigned arbitrarily, and another is designed from formula (12.7).

Calculation of the swirl injector.

The special feature/peculiarity of the work of the swirl injector is the fact that the liquid moves in the injector not over entire its cross section: as a result of the torsion of liquid along the axis/axle of injector appears gas vortex/eddy with the pressure, equal to ambient pressure, i.e., to combustion chamber pressure. The radius of gas vortex/eddy $r_{г.в}$ is lower than the radius of blast nozzle r_{c}. Consequently, liquid flows out behind blast nozzle through the ring cross-section with the area

$$f_{ж}=\pi\left(r_{c}^{2}-r_{г.в}^{2}\right).$$

The speed of the liquid, which escapes from the swirl injector, can be decomposed on the axial component W_a and tangential component W_u.

Component W_a determines fluid flow rate through the injector, while constituting W_u - the torsion of liquid by injector.

Page 202.

Consequently, volumetric fluid flow rate through the nozzle of the swirl injector

$$\dot{v}=W_a f_ж=W_a\pi(r_c^2-r_{г.в}^2)$$

or

$$\dot{v}=W_a\varphi\pi r_c^2,$$

where φ - coefficient of clear opening, determined according to the formula

$$\varphi=1-\frac{r_{г.в}^2}{r_c^2}.$$

Mass fluid flow rate through the nozzle of the swirl injector can be determined according to the formula which in appearance is analogous the flow equation through jet injector (12.5):

$$\dot{m}_ф=\mu f_c\sqrt{2\Delta p_ф\varrho},$$

whence

$$f_c=\frac{\dot{m}_ф}{\mu\sqrt{2\Delta p_ф\varrho}}. \qquad (12.9)$$

The coefficient of the flow rate of the swirl injector μ depends on the coefficient of clear opening φ, i.e., from the clear area $f_{ж}$.

The quality of the atomization of liquid by the swirl injector affects the degree of the torsion of liquid, which determines the angle of spray cone 2a; with its increase is improved the atomization of liquid, but simultaneously increase/grow the necessary sizes/dimensions of injector.

The values 2a, φ and μ of the swirl injector depend on its geometric characteristic, which is the complex, which links the basic dimensions of injector. The geometric characteristic of the swirl injector (Fig. 12.13) they designate by letter A and they determine from the following formulas:

a) for the injector with one tangential opening/aperture

$$A=\frac{R_{вх} r_c}{r_{вх}^2}; \quad (12.10)$$

b) for the injector with a number of tangential openings/apertures i

$$A=\frac{R_{вх} r_c}{i r_{вх}^2}; \quad (12.11)$$

c) for the auger injector

$$A=\frac{\pi R_{вх} r_c}{i f_i}\sin\beta, \quad (12.12)$$

where $R_{вх}$ - the mean radius of channel;

f_1 - flow area of one channel;

i - number of channels (or the approaches of worm screw);

β - helix angle.

With the increase of value A the coefficients φ and μ are decreased, and angle 2a increases/grows.

In the extreme case (with A→∞) we have

$$\varphi \to 0 \overset{(1)}{\text{и}} \mu \to 0.$$

Key: (1). and.

Page 203.

The graph/diagram of the dependence μ and 2a on the geometric characteristic is depicted in Fig. 12.14.

Account to the viscosity of liquid. Relationships/ratios examined above are valid for the ideal fluid. The course of ideal fluid in the swirl injector obeys the law the conservation of angular momentum, since the moment of the external forces, which effect on

the liquid in the chamber/camera of the torsion of injector, is equal to zero.

In the real liquid due to the presence of the viscous forces appear the frictional forces. Their effect/action leads to the fact that the moment of momentum at the nozzle entry proves to be less than in the initial part of the chamber/camera of the torsion of injector, i.e., due to the frictional forces is decreased the degree of the torsion of liquid and as a result, increases/grows the coefficient of flow rate and is decreased the angle of the atomization of liquid.

For the account to the viscosity of liquid instead of the geometric characteristic of injector A is used lumped characteristic $A_{эк}$, determined according to the formula

$$A_{эк}=\frac{R_{вх}r_c}{ir_{вх}^2+\frac{\lambda}{2}R_{вх}(R_{вх}-r_c)}. \qquad (12.13)$$

The coefficient of friction λ for the entry conditions into the injector is designed from the equation

$$\lg\lambda=\frac{25,8}{(\lg Re_{вх})^{2,58}}-2, \qquad (12.14)$$

where $Re_{вх}$ - Reynolds number, determined for the conditions for the entry into the injector.

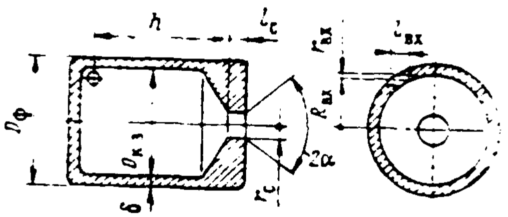


Fig. 12.13.

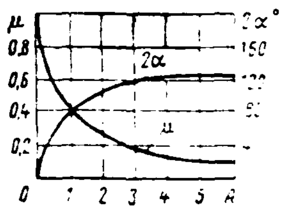


Fig. 12.14.

Fig. 12.13. Tangential injector (on drawing are represented basic geometric dimensions of injector).

Fig. 12.14. Dependence of coefficient of flow rate μ and angle of spray cone 2α on geometric characteristic A.

Page 204.

Value $R_{вх}$ is determined according to the expression

$$Re_{вх}=\frac{4\dot{m}_ф}{\mu_{к.в}\pi d_{вх}\sqrt{i}}, \qquad (12.15)$$

where $\mu_{к.в}$ - kinematic viscosity coefficient of liquid at the entry into the injector.

Order of calculation. The swirl injector is designed in the following sequence.

1. We are assigned by injector pressure drop $\Delta p_ф$ (see pg. 199).

2. We select angle of atomization 2α in limits of 2α=30-120° (most frequently 90-120°).

3. Knowing angle 2α, on graphs, depicted in Fig. 12.14, we determine geometric characteristic A coefficient of flow rate μ.

4. Using equation (12.9), we design sectional area of blast nozzle f_c and then diameter of nozzle according to formula

$$d_c=\sqrt{\frac{4}{\pi}f_c}.$$

5. We select sizes/dimensions of injector.

A number of tangential openings/apertures or approaches of worm screw i is usually taken by the equal to 2-4. An increase in the number indicated improves the distribution of specific weight flow according to the perimeter of the circumference of liquid jet.

Relation $R_{вх}/r_c$ they take as the equal to approximately/exemplarily 2.5.

Using equation (12.11), we determine by the selected values of i and $R_{вх}/r_c$ radius $r_{вх}$:

$$r_{вх}=\sqrt{\frac{R_{вх}r_c}{iA}}.$$

Usually radius $r_{вх}$ is selected in limits $r_{вх}=0{,}25 \div 1$ мм

6. From formulas (12.15) and (12.14) we design coefficient of friction λ, and then in equation (12.13) - lumped characteristic of injector $A_{эк}$. If characteristics A and $A_{эк}$ differ not more than to [illegible], then calculation is finished, in this case sizes/dimensions r_c, $R_{вх}$ and $r_{вх}$ of first approximation they accept as the final ones.

If the disagreement of characteristics A and $A_{эк}$ large, then we take for the basis value $A_{эк}$, obtained in the first approximation, and on the graph/curve, depicted in Fig. 12.14, we determine the coefficient of flow rate μ taking into account to viscosity, and then sizes/dimensions r_c, $R_{вх}$ and $r_{вх}$ in the second approximation/approach; from them we design characteristic $A_{эк}$ in the second approximation/approach. Usually the disagreement of values $A_{эк}$, of those obtained with the first and second approximations/approaches, is insignificant, so that sizes/dimensions r_c, $R_{вх}$ and $r_{вх}$, obtained in the second approximation/approach, can be accepted for the final ones.

7. Knowing r_c, $R_{вх}$ and $r_{вх}$, we select remaining sizes/dimensions of injector (see Fig. 12.13)

$$l_{вх}=(1{,}5 \div 3)\,d_{вх};\; l_c=(0{,}25 \div 1{,}0)\,d_c.$$

Page 205.

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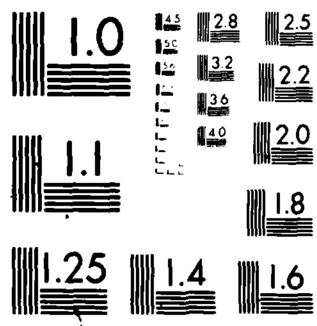

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The height/altitude (length) cf injectcr h we accept:

a) $h=R_{вх}$ and more - fcr the tangential injector;

b) $^1/_4-^1/_3$ the space cf channel or are mcre - for the auger injectoz. The diameter cf the chamber/camera cf torsicn is equal to

$$D_{к.з}=2(R_{вх}+r_{вх}).$$

The outside diameter of injectcr is equal to

$$D_{ф}=D_{к.з}+2\delta,$$

where δ - thickness cf the chamber wall of torsion.

Sizes/dimensions δ and $l_{вх}$ are ccnnected. Usually are selected by δ=1.5 mm.

Special features/peculiarities cf the heads cf afterburners.

Depending on the state of aggregation of the prcpellant component, introduced inside afterburner, injectcrs subdivide into the liquid ones, the gas cnes and the gas-liquid. Gas-liquid are called the duplex-fuel nozzles into which one component enters in the liquid state, and ancther - in the gaseous.

Generator gas inside afterburner is introduced through the jet injectors.

The head of chamber/camera ZhRD, that work on the diagram " gas-liquid", can be represented itself grid/cascade with the radial and ring cross connections, moreover windows are the jet injectors of generator gas, and the injectors of liquid component are placed in the units of cross connections.

The pressure differential on the jet injectors of generator gas is small, and pressure in afterburner large; therefore the outflow of gas from the injector subcritical.

§ 12.7. Selection of volume and relative area of combustion chambers (afterburnings).

The combustion chamber volume (afterburning) must provide the necessary retention time of propellant components in it, moreover dimensions and mass of chamber/camera must be low.

The combustion chamber volume is designed from its reduced length $l_{пр}$ and conditional retention time of gas in the combustion chamber $\tau_{усл}$.

The given (or characteristic) length combustion chamber calls the ratio of its volume to area of the critical cross section

$$l_{\text{пр}}=\frac{V_{\text{к}}}{f_{\text{кр}}}. \qquad (12.16)$$

Page 206.

Time $\tau_{\text{усл}}$ can be obtained, after dividing the mass of gas in the combustion chamber into its flow rate per second:

$$\tau_{\text{усл}}=\frac{m_{\text{газ}}}{\dot{m}};$$

disregarding the volume of liquid propellant components in the combustion chamber and conditionally considering that gas density in entire its volume one and the same and is equal to $\varrho_{\text{к}}$, we obtain

$$\tau_{\text{усл}}=\frac{V_{\text{к}}\varrho_{\text{к}}}{\dot{m}}.$$

Let us substitute in the latter/last equation expression $\varrho_{\text{к}}$ from formula (4.4) and will consider relationships/ratios (4.14) and (12.16). Then

$$\tau_{\text{усл}}=\frac{\beta}{RT_{\text{к}}}l_{\text{пр}}. \qquad (12.17)$$

For the prescribed/assigned components of propellant and construction/design of head, which is determining the quality of carburetion, relation $\beta/RT_{\text{к}}$ can be considered constant. Consequently, the conventional retention time of gas in the combustion chamber and its reduced length are found in the directly proportional dependence.

Values $\tau_{\text{усл}}$ and $l_{\text{пр}}$ are determined mainly by fuel/propellant, construction/design of head and by type of diagram of ZhRD; for the

majority of engines $\tau_{ycл}=(1,5\div5,0)\,10^{-3}$ s and $l_{пр}=1,0\div3,5$ m. Smaller value $\tau_{ycл}$ they correspond to chambers/cameras with large pressure $p_{к}$. With an increase in the reduced length of combustion chamber increases/grows the specific impulse, but simultaneously are increased the sizes/dimensions of chamber/camera, which complicates its cooling.

For tentative calculations the reduced length of the combustion chambers of ZhRD, which work according to the diagram "liquid-to-liquid" on the fuels/propellants O_2+kerosene, F_2+NH_3 and O_2+H_2, it is possible to accept 1.5-2.5; 1-1.5-1 m with respect [to 4], [17].

In ZhRD with the afterburning of generator gas the part of the propellant components burns preliminarily in the gas generator; therefore the necessary reduced length of their afterburner is 1.3-1.8 times less than for the combustion chambers of ZhRD, which work on the diagram "liquid-to-liquid".

When selecting of the optimal relationship/ratio between length and diameter of combustion chamber (afterburning) is used its relative area $\bar{f}_{к}$.

Besides the shortcomings, noted into § 12.2, with the decrease

of value $\bar{f}_к$ additionally becomes complicated the organization of the effective atomization of propellant components due to reduction in area of surface, over which are placed the injectors.

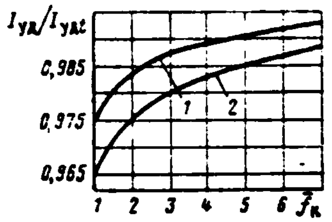

Fig. 12.15. Dependence of relation $I_{уд}/I_{удt}$ on value $\bar{f}_к$ when $\varepsilon_c=100$ (curve 1) and $\varepsilon_c=10$ (curve 2).

Page 207.

Therefore with the decrease of relative area $\bar{f}_к$ the specific impulse of chamber/camera descends (Fig. 12.15), which is noticeable when $\bar{f}_к<3$ (especially when $\bar{f}_к\approx1$). The effect of the relative area of combustion chamber (afterturning) on the specific impulse when $\bar{f}_к>3$ can be disregarded/neglected especially with the high expansion ratio of gas ε_c.

Some advantages of the selection of low relative area $\bar{f}_к$ include the decrease of the mass of chamber/camera and the facilitation of its cooling (is decreased the necessary thickness of the combustion chamber walls and its surface, which it is necessary to cool).

Relative area $\bar{f}_к$ can be determined on the selected specific weight flow of combustion chamber from equation (12.2), which taking

into account formulas (4.14) and (12.1) can be written in the following form:

$$r=\frac{p_\kappa}{\bar{f}_\kappa\beta}. \qquad (12.18)$$

Since for this fuel complex β can be considered constant, then with an increase in pressure p_κ the specific weight flow of combustion chamber also increases/grows.

Relation r/p_κ is called relative specific weight flow and designate r_p, i.e.

$$r_p=\frac{r}{p_\kappa}, \qquad (12.19)$$

or taking into account equation (12.18)

$$r_p=\frac{1}{\bar{f}_\kappa\beta}. \qquad (12.20)$$

If for the fuels/propellants used complex β is equal to 1700-2400 r•s/kg [~~170-240 kG•s/kg], then when $\bar{f}_\kappa=2\div6$ relative specific weight flow constitutes (0.1-0.2) 10^{-3} kg/(N•s) [~~(1-2) of 10^{-3} kg/(kg•s)] [17].

To the value $\bar{f}_\kappa$ indicated for the cylindrical chambers/cameras corresponds the ratio of the length of combustion chamber to the diameter of its cylindrical part l_κ/d_κ, equal, 1.0-1.5.

Page 207.

Chapter XIII.

FEED SYSTEMS OF LIQUID PROPELLANT COMPONENTS.

Construction/design of ZhRD to a considerable degree depends on the system, with the aid of which is created a pressure, necessary for supplying the liquid propellant components into the chamber/camera. In ZhRD are used in essence pressurization and pump feed systems.

In the pressure feed system the pressure in the fuel tanks is more than in the engine chamber. DU with the pressururized-propellant feed are simple and reliable, but they have large mass ratio of tanks.

Page 208.

In ZhRD with the pump system of component the fuels/propellants

from the tanks are supplied into barrel, given by gas turbine. Turbopump aggregates complicate construction/design ZhRD and its operation. In the presence of TNA appear the problems, connected with the work of bearings, packings/seals, by rotor balancing.

However, pump feed system possesses the series/number of advantages.

1. Necessary pressure in tanks is small - usually 2-4 bars [≈2-4 kgf/cm²]. Therefore DU with the pump feed system have substantially smaller specific mass, than DU with the pressurization system.

2. Engine power rating can be comparatively easily changed, changing number of revolutions of shaft of TNA.

3. Is possible creation of large pressures of propellant components with relatively low dimensions and mass TNA.

Propellant components into the engine chamber can be supplied by jet pumps, or ejectors (Fig. 13.1). This system was for the first time proposed by K. E. Tsiolkovskiy in 1914. In the jet pump the pressure of liquid propellant component is raised (as a result of ejection) by the supplied (carrying) gas. This pump consumes greater gas flow rate per second, than turbine of TNA, but its

construction/design is simpler and reliability is above as a result of the absence of mobile parts. Furthermore, for the work of jet pump is required low pressure in the tanks.

The jet pump, lifting working body of which is liquid, but not gas, usually is used as the auxiliary unit, which makes it possible to improve characteristics of TNA, namely to raise the permissible number of revolutions of shaft of TNA as a result of an increase in the pressure at the entry into the centrifugal pumps (see §13.10).

In the rocket vehicles with large total impulse I_{Σ}, which include the boost-glide vehicles (space aircraft), use pump feed of propellant components, while in the rocket vehicles with low I_{Σ} (space vehicles and ships) - pressurization.

During the design of concrete rocket vehicle is selected that feed system, which with the prescribed/assigned characteristic velocity or total impulse I_{Σ} provides the smaller initial mass of vehicle and greater relation $I_{\Sigma}/m_{\text{ДУ}}$. Relations $I_{\Sigma}/m_{\text{ДУ}}$ must be compared at optimal pressures $p_{\text{к}}$ for the pressurization and pump feed system.

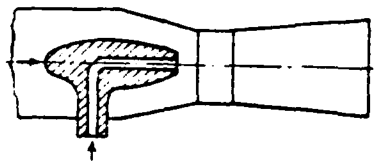

Fig. 13.1. Schematic of jet pump (ejector).

Page 209.

Pressure feed system it is expedient to use at comparatively low pressures p_K - usually 20-25 bars [≈20-25 kgf/cm²] -, also, for the propellant components with the increased density ($F_2+N_2H_4$; F_2+NH_3; $OF_2+B_2H_6$, etc.) and also with the relatively short operating time of engine. In certain cases are selected the pressure feed system in view of its large simplicity and the reliability, in spite of the lower value of the characteristic velocity of rocket vehicle.

Into the feed systems of the components of the propellant of engine installations enter different valves, chokes/throttles and regulators from description of which it is expedient to begin the examination of the systems indicated.

§13.1. Valves.

Valves are intended for discovery/opening or overlapping those

or ine of the mains of engine installation and in the majority of the cases have two operating positions: "it is opened" and "closed".

To the valves DU present the following basic requirements:

1) the high reliability of operation;

2) low power necessary for the operation;

3) small hydraulic resistance;

4) complete airtightness in the closed position.

Fig. 13.2 depicts the valve of the simplest construction/design, which consists of the housing, disc valve and spring. In the absence of pressure or at a low pressure at the entry into the valve the spring forces disc valve against saddle, in this case in the place of contact is developed the necessary specific pressure, which ensures the airtightness of valve in the closed position. With pressure rise at the entry into the valve, for example during engine starting, grows the force, which effects on the disc and equal to the product of pressure on the flow passage cross-sectional area of intake connecting pipe 1 with diameter d. When this force exceeds force of compression of spring, disc will move away from the saddle and valve

is opened/disclosed. This valve is called reverse/inverse. The effect/action of check valve lies in the fact that with an increase in the line pressure at the output/yield from it (in connecting pipe 11) or with lowering in the pressure in the connecting pipe 1 valve is closed, without allowing/assuming the flow of liquid or gas in the opposite direction.

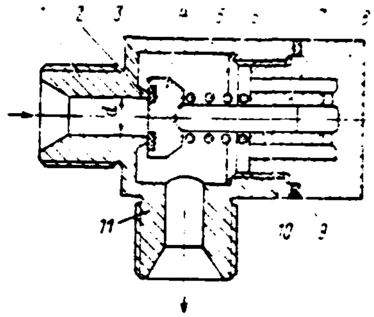

Fig. 13.2. The check valve: 1 - intake connecting pipe; 2 - saddle; 3 - ferrule; 4 - disc; 5 - housing; 6 - spring; 7 - being guided; 8 - cover/cap; 9 - ply; 10 stock/rod; 11 - output connecting pipe.

Page 210.

Valves can be classified according to the following signs.

1. On kind of medium: liquid valves (among them it is necessary to isolate valves for cryogenic components of propellant and working medium/propellants) and valves of hot and cold gas.

2. On normal position of valve: "it is normally opened", "normally-closed" and "normally partially opened". Normal is called this position of the valve in which it is located in the absence of any effect on it from without (i.e. without the pressure, without the supply of control current, etc.). Check valve examined above is the

example of the normally-closed valve, and also the example of the unguided valve. The position of check valves is determined by the pressure of component of propellant or working medium/propellant at the entry in them.

The majorities of valves are controlled, i.e., they have one or another the drive which at the required moment of time can forcedly (independent of line pressure on which is established/installed the valve) to create the force under effect/action of which the valve is opened/disclosed or is closed.

3. According to type of drive:

a) valves with electric drive; force, necessary for opening either closing of valve, it is created by electromagnet, or solenoid;

b) valves with pneumatic drive; such valves have control cavity, into which at required moment of time is supplied any compressed gas (air, nitrogen or helium);

c) valves with hydraulic drive; into control cavity of such valves is supplied under pressure any with liquid; for this purpose frequently select/take small part of one of basic components propellants (usually fuel) from main at output/yield from pump;

d) valves with the pyrotechnic drive (pyrotechnic valves); they are opened/disclosed or are closed by the force of pressure of the gases, which are generated with the combustion of a small quantity of pyrotechnic charge in the control cavity.

4. In number of operations: valves of one-time and repeated operation.

The valves of one-time operation include the pyrotechnic valves and the membranes/diaphragms. Are distinguished the membranes/diaphragms of the free breach/inrush (burst open by the pressure of propellant component itself) and the membrane/diaphragm of forced breach/inrush.

Valves with electrical, pneumatic or hydraulic drive are the valves of repeated operation. After the cutoff/disconnection of feed/supply of electromagnet or compression release from the control cavity the spring returns valve to the initial position, after which it can again operate/wear during the supplying of control signal.

Page 211.

5. According to designation/purpose: reverse/inverse, starting/launching, cutoff, main fuel, drainage, drain-reserve, filling and draining, etc.

Starting/launching call the normally-closed valves during discovery/opening of which the propellant component or working body enters the main on which they are established/installed.

Cutoff are the normally open valves, which rapidly cover main and ceasing thereby the entrance or component of propellant or working medium/propellant into the aggregate, before which they are established/installed.

In a number of cases of functioning the starting/launching and cutoff valves it can perform one valve. Such, in particular, are the main fuel valves, adjusted on the basic mains of components of propellant (from the tanks to the engine chamber).

Drain valves are intended for drainage (i.e. letting out, throw-out into the environment) of the components of propellant, their vapors and so forth. The drain valves of the system of cooling mains DU with the cryogenic propellant components are opened/disclosed in the process of cooling (in this case valves on the entry into the chamber/camera and ZhGG are closed) and are closed

directly before engine starting.

Drain-reserve valves (DRK) are established/installed to the tanks, and also on the main for the exception/elimination of the excess of the prescribed/assigned pressure in them.

As the starting/launching, cutoff, main and drain valves it is possible to use valves with pneumatic, hydraulic and pyrotechnic drive.

Drain-reserve valves are actually the check valves, which by of trial and error of the corresponding spring adjust to the specific pressure (pressure of adjustment); with its excess the valve is opened/disclosed, and overpressure bronzes from the tank or the main, which prevents their destruction or other inadmissible consequences.

The construction/design of the membrane/diaphragm of free breach/inrush (Fig. 13.3) and its fastening in the joint must provide the guaranteed strength and airtightness to the required pressure, and with the excess of design pressure the membrane/diaphragm must be torn. For facilitating of explosion and decrease of spread the bursting pressures on the membrane/diaphragm make cut, usually in the form of arc. With this form of cut is eliminated the possibility of the breakaway of the lobe/lug of the membrane/diaphragm: the

membrane/diaphragm bursts open on the out, moreover formed the lobe/lug of the membrane/diaphragm is unbent by the fluid flow or gas.

Valve with electric drive more frequently is established/installed in the mains of gas; such valves are called the electrical air operated valves (EPK).

Page 212.

Let us examine operating principle EPK with drainage (Fig. 13.4). During the supplying of direct current to the coil appears the magnetic field, which is strengthened by yoke (core) from soft iron. Armature is attracted/tightened to the yoke, and the emergent effort/force through the stock/rod opens/discloses foot valve and closes overhead valve, in this case the gas can flow/occur/last through the valve. During de-energizing of coil, i.e., after the disappearance of magnetic field, the spring returns the system of valves to the upper position, so that the duct of the gas through the valve ceases. Since overhead valve is opened, then gas from the cavity or the main bronzes through the drain holes into the environment.

EPK with drainage extensively use in the mains of valve control

with the pneumatic drive (Fig. 13.5). Valve is closed by the force of the compressed spring and by the pressure of propellant component on the disc of valve. During the supplying of the controlling/guiding gas the force of pressure on the piston opens/discloses valve and holds it in the open position. With the compression release from the control cavity, for example by operation EPK with drainage, the valve is again closed. In this valve the control cavity is isolated from the liquid propellant component by the cavity, which is communicated with the environment, but spring does not contact with the propellant component. The control cavity on the piston and the cavity of liquid component on the stock/rod are hermetically sealed by rubber gaskets.

The example of main fuel valves are the valves which in the normal position are closed under the effect/action of spring strength, and with the work of engine are opened/disclosed under the effect/action of the increasing pressure at their entry and are closed during the supplying the gas pressures in the control cavity.

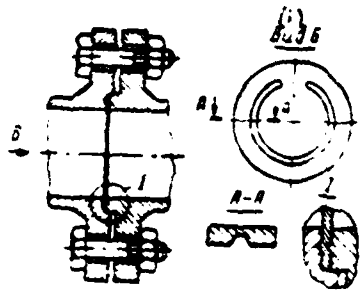

Fig. 13.3.

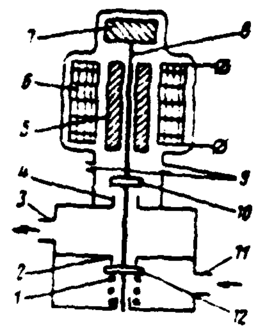

Fig. 13.4.

Fig. 13.3. Membrane/diaphragm of free breach/inrush.

Fig. 13.4. Schematic of electrical air operated valve (EPK): 1 - spring; 2 - lower saddle; 3 - output connecting pipe; 4 - upper saddle; 5 - yoke of electromagnet; 6 - magnet coil; 7 - armature; 8 - stock/rod; 9 - drain holes; 10 - overhead valve; 11 - intake connecting pipe; 12 - foot valve.

Page 213.

Valves with the pyrotechnic drive (pyrotechnic valves) are fulfilled by those by normally opered and normally-closed.

Normally opered by cutoff pyrotechnic valves (Fig. 13.6) consists of housing with the saddle, valve and explosive charge.

valve is held in the open position with the aid of the membrane/diaphragm; for this purpose it is possible to use also a tag. The current, flowing through the hot wire, heats it and ignites the inflammable pyrotechnic composition, placed around the filament. The formed flame priming charge of the explosive charge (Fig. 13.7), the combustion products of which tear the bottom of housing and act on valve spindle, sharply moving it. The disc of valve is wedged in the saddle, hermetically sealing valve after its operation.

The normally-closed starting/launching pyrotechnic valve (Fig. 13.8) is the membrane/diaphragm or forced breach/inrush. Such valves usually are established/installed in the entry in the main of engine. Valve operates/wears during the supplying of current to explosive charge 5. The combustion products of the charge of explosive charge act on diaphragm 4, in this case it is deflected and moves knife 2, which shears tags 3 and cuts through the membrane/diaphragm 1. The membrane/diaphragm is urbent under the pressure of the propellant component which enters the main of engine.

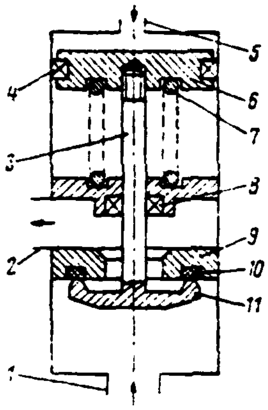

Fig. 13.5.

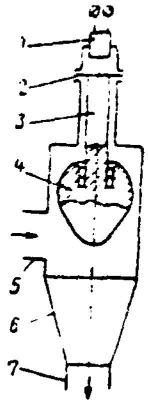

Fig. 13.6.

Fig. 13.5. Schematic of valve with pneumatic drive: 1 - intake connecting pipe; 2 - output connecting pipe; 3 - stock/rod; 4 - piston packing; 5 - connecting pipe of delivery of controlling/guiding pressure; 6 - piston; 7 - spring; 8 - packing/seal of stock/rod; 9 - saddle; 10 - ferrule; 11 - disc.

Fig. 13.6. Diagram of normally open cutoff pyrotechnic valve: 1 - explosive charge; 2 - membrane/diaphragm; 3 - stock/rod; 4 - disc of valve; 5 - intake connecting pipe; 6 - saddle; 7 - output connecting pipe.

Page 214.

Such valves in contrast to the membrane/diaphragm of free

breach/inrush operate/wear at the required moment of time during the supplying of the corresponding command/crew, what is their great advantage.

§13.2. Regulators, chokes/throttles and sensors.

To the regulators and the chokes/throttles carry such aggregates of automation with the aid of which change the parameters DU or rocket vehicle as a whole on predetermined program or flowing signals of control-system equipment. With the aid of some regulators in the known limits are maintained the pressure or the flow of gas or liquid. Regulators DU are, for example, pressure reducers of gas. Regulators and chokes/throttles must possess operating speed, i.e., time from the delivery of the command on them to its accomplishing must be low.

Pressure reducers of gas. When must be constant pressure in the tank, from which continuously is expended/consumed liquid component of the propellant (or liquid it is working body), so that the volume of gas cushion/pad increases/grows, is used the pressure reducer of gas, adjusted between the gas container high-pressure and the tank. Pressure reducer provides with the low error (to 0.15o/o) the prescribed/assigned gas pressure at the output/yield from the reductor despite the fact that the pressure at the entry into it

continuously falls as a result of the outflow of gas from the tank/balloon. The gas pressure in the tank/balloon in the beginning of the work of engine must be substantially higher than the required pressure gas in the tank. Toward the end of the work of engine the gas pressure on the entry into the reductor must to certain value exceed pressure at the output//yield from the reductor and, consequently, also pressure in the tank. Gas pressure drop on the reductor is necessary, since precisely via different degree of throttling/choking (breaking) gas is provided the work of reductor. In proportion to decompression of gas in the tank/balloon the degree of throttling/choking continuously is decreased.

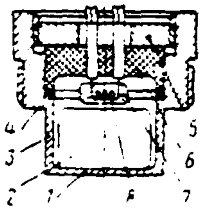

Fig. 13.7.

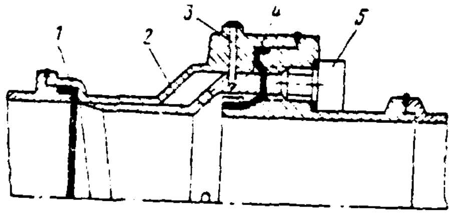

Fig. 13.8.

Fig. 13.7. Explosive charge: 1 - bottom; 2 - charge; 3 - housing; 4 - plv; 5 - nut; 6 - insulator; 7 - inflammable mixture; 8 - hot wire.

Fig. 13.8. Normally-closed starting/launching pyrotechnic valve (membrane/diaphragm of forced breach/inrush): 1 - membrane/diaphragm; 2 - knife; 3 - tag; 4 - diaphragm; 5 - explosive charge.

Page 215.

Throttling/choking gas in the reductor is accomplished/realized in transit through the throttling cross section, which is the narrow annular slot between the valve and the saddle. With the decrease of size of slit the degree of throttling/choking gas is increased, and its pressure after the throttling opening/aperture is decreased, and vice versa.

Depending on that, in which direction relative to the direction

of the motion of gas is opened/disclosed the valve of reductors, them are subdivided into the reductors of the direct and reverse/inverse effect/action. In the reductor of the direct effect/action the valve is opened/disclosed in the direction of the flow of gas, while in back-action reductor - in the opposite direction. More frequent in DU are used pressure reducers of back-action.

In the pressure reducer of back-action (Fig. 13.9) it is possible to isolate three cavities: the cavity of high and low pressure and submembrane cavity. If adjusting screw 1 is established/installed to such position in which on the membrane/diaphragm 3 does not act the force or this force is low, then the valve of reductor under the effect of pressure in the cavity of high pressure and force of compression of spring 6 is closed. So that the reductor would enter the effect/action, i.e., it provided the constant prescribed/assigned pressure at the output/yield from it with the decrease of pressure at the entry, it is necessary with the aid of the adjusting screw to adjust spring 2. Mobile system the membrane/diaphragm - stock/rod - valve at each given moment of the work of reductor occupies such position in which the force of spring 2 is balanced by the sum of three forces, which effect in the opposite direction:

a) the force of pressure of gas on the membrane/diaphragm in

low-pressure cavity;

b) the force of spring 6;

c) the force of pressure of gas from the side of high-pressure cavity.

The selection of this compression springs by the adjusting screw, with which at the output/yield from the reductor is provided the prescribed/assigned gas pressure, call the adjustment of reductor.

In the process of the work of pressure reducer of gas in high-pressure cavity continuously is decreased, which leads to the disturbance/breakdown of equilibrium of the forces, which effect on the mobile system; therefore it is continuously moved upward, increasing area throttling openings/apertures and providing the constant gas pressure at the output/yield from the reductor by decreasing the degree of throttling/choking.

Page 216.

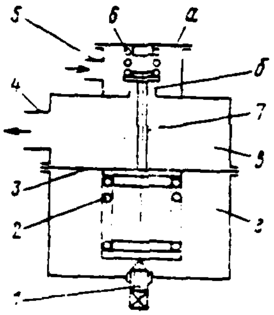

Fig. 13.9.

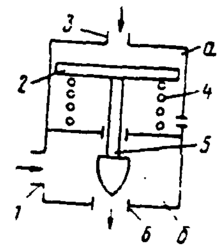

Fig. 13.10.

Fig. 13.9. Diagram of pressure reducer of gas: 1 - adjusting screw; 2, 6 - spring; 3 - membrane/diaphragm; 4 - output connecting pipe; 5 - intake connecting pipe; 7 - stock/rod; a) high-pressure cavity; b) throttling opening/aperture; c) low-pressure cavity; d) submembrane cavity.

Fig. 13.10. Schematic of choke/throttle with shaped needle: 1 - intake connecting pipe; 2 - piston; 3 - connecting pipe of delivery of controlling/guiding pressure; 4 - spring; 5 - stock/rod with shaped needle; 6 - output connecting pipe: a) cavity of propellant component; b) cavity of controlling/guiding pressure.

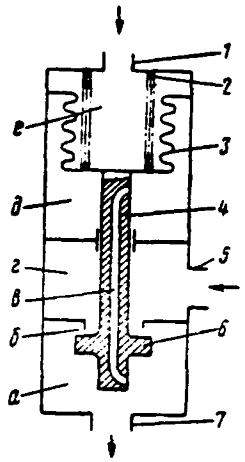

Fig. 13.11.

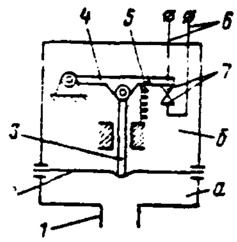

Fig. 13.12.

Fig. 13.11. Regulator of pressure constancy of supply of propellant components: 1 - connecting pipe of delivery of controlling/guiding gas; 2 - spring; 3 - bellows; 4 - stock/rod; 5 - intake connecting pipe; 6 - valve; 7 - output of connector; a) output cavity; b) throttling cross section; c) drilling in stock/rod; d) intake cavity; e) cavity, connected with output cavity; f) cavity of controlling/guiding gas.

Fig. 13.12. Diagram of signal indicator (relay) of pressure: 1 - connecting pipe of delivery of pressure of component of propellant or gas; 2 - membrane/diaphragm; 3 - push rod; 4 - lever; 5 - spring; 6 - electrical chains; 7 - contacts a) dynamic cavity; b) static cavity.

Page 217.

If on any reason the gas pressure in low-pressure cavity deviates from the prescribed/assigned value, then disturbs balance itself of the forces, which affect on the mobile system. At the same time in the case of decreasing the pressure at the output/yield from the reductor is decreased the force of pressure of gas on the membrane/diaphragm and force of compression of spring from the side of adjusting screw moves mobile system upward, which leads to the recovery of required pressure, and vice versa; consequently, the required pressure at the output/yield from the reductor it is maintained by constant automatically.

The adjustment of reductor can be changed both before engine testing and with its work. In the latter case to the adjusting screw it must be connected any lead (for example, electrical).

Regulators and chokes/throttles. Into composition DU enter the actuating elements of the systems which change in certain range, and also maintain with constants the fluid flow rate or gas, coefficient $\varkappa$ and other parameters.

The fluid flow rate or gas can be changed with choke/throttle with the shaped needle (Fig. 13.10). With the displacement/movement of needle changes flow area between the saddle and the valve and, consequently, also fluid flow rate or gas. Needle can be moved under the gas pressure or liquid, which effects on the piston from the side of the control cavity. Using one or another the profile/airfoil of needle, it is possible to obtain the necessary change in the fluid flow rate or gas.

For the safeguard of condition $\varkappa$=cosnt or change of coefficient within some limits install choke/throttle the systems SOB in main of one of the propellant components, which during the supplying of control signal changes to the necessary degree the expenditure/consumption of component.

FOOTNOTE [1]. SOB - system of the synchronous emptying of tanks. ENDFOOTNOTE.

The regulator of the pressure constancy of the supply of component of propellant (Fig. 13.11) provides its constant pressure in the output cavity at an invariable pressure of gas in the control cavity. The component through connecting pipe 5 enters the intake cavity d and through the throttle cross section b - into the output cavity a. Regulator has the mobile system, which consists of bellows

3, stock/rod 4 with drilling c valve 6. Drilling c connects cavity a with cavity e.

Spring 2, bellows 3 the area of valve 6 can be selected so that in the case of changing the pressure in cavity a, i.e., pressure at the output/yield from the regulator, disturbs balance itself of the forces, which effect on the mobile system, and it is moved toward of decrease or increase in the throttling cross section b, as a result of which the prescribed/assigned pressure in cavity a is restored.

Sensors of systems DU. In systems DU are included the sensors, which supply into the control system the signal about the achievement of the upper or lower allowed value of the parameter (pressure, temperature, the number of revolutions of turbine, etc.) or the signal, proportional to the value of the measured parameter.

Page 218.

The signal indicator (relay) of pressure (Fig. 13.12) has two or three pairs of the contacts which are connected with the aid of the transmission gear through the push rod with the membrane/diaphragm. Under the pressure of liquid or gas the membrane/diaphragm is deflected, which leads to the closing of contacts (for the signal indicator with the normally open contacts) or to breaking of contacts

(for the signal indicator with normally closed contacts). With decompression the contacts respectively are broken or are closed.

§13.3. Feed systems with the gas storage tank of pressure.

The gas storage tank of pressure (AD) consists of tank/balloon with the compressed gas, pressure reducer of gas and valves.

Fig. 13.13 depicts diagram DU of one-time inclusion/connection with AD gas. The tank/balloon AD gas they service high-pressure by compressed gas through valve 3. Engine is included by valve opening 4. Gas passes through the pressure reducer by 5, where its pressure descends to the assigned magnitude, breaks through the membranes/diaphragms 6, are opened/disclosed check valves 7 and it enters tanks, as a result of which the pressure in them increases/grows. Upon reaching of the prescribed/assigned pressure they are closed the contacts of pressure indicator by 8, in this case passes the command/crew to the discovery/opening of main valves 10.

The prescribed/assigned mode/conditions of work ZhRD is provided by the adjustment of reductor 5, and the required fuel component ratio - by tuning disks 11.

Engine is turned off/disconnected by the coverage of valve 4 and

main valves 10.

The membranes/diaphragms 6, adjusted in the conduits/manifolds, which supply gas into the tanks, eliminate the contact of fuel and oxidizer to engine starting, and check valves 7 fulfill the same function during the work of engine.

Usually systems with AD gas supply gas into the tanks continuously during entire operating time of engine. However, such systems can work also in the pulsed operation, in this case is not required the pressure reducer. In this case in the conduit/manifold, which supplies gas from the tank/balloon into the tanks, is installed the valve, and on upper bottom of one of the tanks - two pressure indicators.

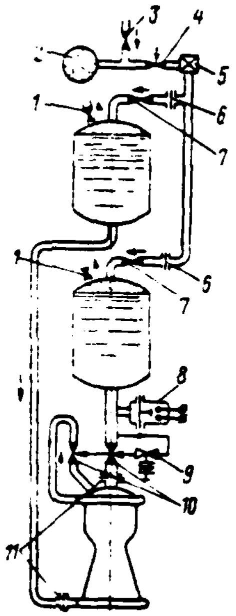

Fig. 13.13. Diagram DU with the gas storage tank of the pressure: 1 - drain-reserve valve; 2 - tank/balloon with the compressed gas; 3 - filling valve; 4 - cutoff valve; 5 - pressure reducer of gas; 6 - burst diafragm; 7 - check valve; 8 - pressure indicator; 9 - valve with the electromagnet; 10 - main valves, which have the system of drive from one controlling/guiding piston; 11 - tuning disks.

Page 219.

One signal indicator it is adjusted on minimally, and another - to the maximum permissible pressure in the tank. With decompression in the tank to that the minimally permitted they are closed the contacts

of pressure indicator and passes command/crew to valve opening. Pressure in the tanks begins to grow, and upon reaching of the maximum permissible value on the closing of contacts of other signal indicator valve is closed and so forth.

Feed systems with AD gas possess high reliability and sufficient simplicity, it is most finished and extensively they are used. For DU with the gas storage tank of pressure is sufficiently simple the safeguard of multiplying of engine and a change in its thrust.

However, systems with the gas storage tank of pressure possess the relatively larger mass of gas and tank/balloon (as a result of the high initial gas pressure) and are used in essence for the engine installations with the small thrust and the low total impulse.

The system, which uses as the pressurization gas air, is called system with the air pressure accumulator (VAD).

In the engine installations with VAD with the preheating (Fig. 13.14) the air, which enters the tanks, is heated by the heat, which is isolated as a result of reacting the burning of liquid fuel and oxygen, which is contained in the air. For the reliable work of the feed system of oxidizer into its tank must not enter unburned fuel. Is possible the use/application of other methods of heating the

displacing gas before its supply into the tanks.

Selection of an initial storage pressure and of the type of pressurization gas. For decreasing the volume of tank/balloon it is desirable to select large initial gas pressure. Furthermore, with an increase in the gas pressure to certain limit - approximately/exemplarily to 350 bars [≈350 kgf/cm²] - is provided gain in the mass of tank/balloon. The greater gas pressure to use inexpediently, since simultaneously with the decrease of the volume of tank/balloon to the noticeable degree is increased its wall thickness. Usually the initial pressure of pressurization gas in the tank/balloon is selected in the limits from 200 bars [≈200 kgf/cm²] to 350 bars [≈350 kgf/cm²]. For the convenience the arrangements/positions in the rocket vehicle in certain cases use not one, but several tanks/balloons with the compressed gas.

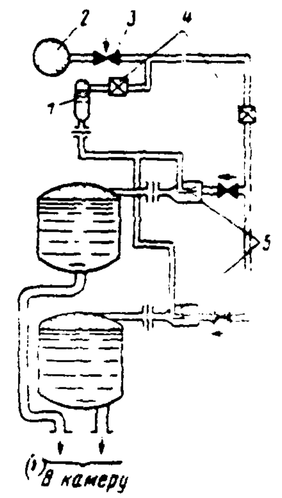

Fig. 13.14. Diagram DU with VAD with the preheating: 1 - small tank with the auxiliary liquid propellant component; 2 - tank/balloon with the compressed air; 3 - cutoff valve; 4 - pressure reducer of gas; 5 - preheating chamber of air.

Key: To chamber.

Page 220.

As the displacing gas of the pressure accumulators serve air, nitrogen, helium or hydrogen. If propellant components can react with oxygen, which are contained in the air, and is absent the separating device between the components of propellant and the compressed gas, then instead of the air (cheapest working medium/propellant) are used the inert gases: nitrogen or helium.

During the displacement of liquid cryogenic propellant components with the aid of air or nitrogen the latter are cooled, are condensed and are dissolved in the fuel/propellant, in consequence of which the flow of the displacing gas is necessary to increase. Therefore cryogenic components it is profitable to displace by helium, which has lowest possible boiling point. Helium it is profitable to use also for the displacement of noncryogenic propellant components, since at one and the same volume of tank/balloon and pressure in it, as a result of the less density of helium, its necessary mass is approximately/exemplarily 7 times lower than the mass of nitrogen.

The additional advantage of helium over air and by nitrogen lies in the fact that the temperature of helium during the throttling/choking in the pressure reducer increases/grows, while for air and nitrogen it is decreased. Therefore utilization of helium instead of nitrogen or air gives a considerable reduction/descent in the mass of feed system. Some shortcomings in helium are its high cost/value and great tendency toward the leakages.

§13.4. Feed systems with the liquid and solid-propellant gas generators.

In the pressure feed systems are used mono- and di-component

ZhGG; is possible the utilization of three-component ZhGG.

The displacing gas in DU with one-component ZhGG is formed as a result of the decomposition of additional liquid component of the propellant (for example, peroxide of hydrogen H_2O_2 or hydrazine N_2H_4), supplied to the gas generator from the special small tank.

Fig. 13.15 shows the schematic of engine installation with di-component ZhGG, established/installed on the upper bottoms of tanks. In ZhGG of the fuel tank of component the fuels/propellants are supplied by selecting the tuning disks with the excess of fuel ($\alpha_{ox}=0,3\div0,4$), while in ZhGG of oxidizer tank - with the excess of oxidizer ($\alpha_{ox}=3\div6$), so that the temperature of the displacing gas would not exceed the values, permitted for the material of the walls of tank. Therefore temperature in ZhGG does not exceed 1300-1500°K. The shield, placed in the upper part of each tank, decreases the effect of the stream of combustion products on the surface of propellant components in the tanks.

The displacement of propellant components of both tanks with the aid of one ZhGG is dangerous to those that in one of the tanks occurs the afterburning of combustion products, which can lead to the explosion; for example, combustion products ZhGG, which contain the excess of fuel, after the entrance into the oxidizer tank will burn

out in it.

Page 221.

In the systems with the solid-propellant gas generators the components of fuel/propellant are displaced from the tanks by the combustion products of solid-propellant grain.

Engine installations with TGG (Fig. 13.16) start by the supply of electrical signal to the igniter of the charge, placed in gas generator 2. Generating combustion products tear the membrane/diaphragm 4, which hermetically seals cavity TGG, and they fall into the units of input/introduction 3, placed in the gas cushion/pad of tanks. During the pressure buildup in unit 3 operates/wears check valve, and combustion products begin to enter tank, in this case they move in essence in parallel to the surface of propellant component, exerting only small influence on it. Check valves in units 3 divide the gas cushions/pads of tanks from each other, including after engine cutoff, which is especially necessary for the self-igniting fuels/propellants. Upon reaching of the prescribed/assigned pressure in the tanks they are closed the contacts of pressure indicator by 5, as a result of which passes the command/crew to the discovery/opening of main valves of engine.

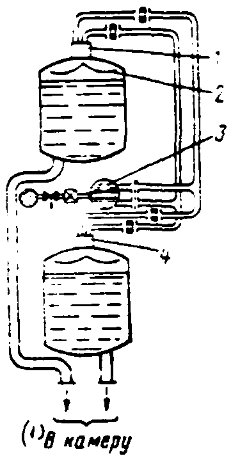

Fig. 13.15.

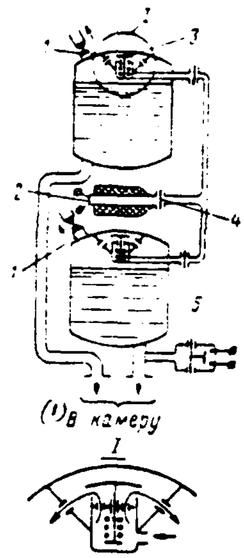

Fig. 13.16.

Fig. 13.15. Diagram DU with two-component ZhGG: 1 - ZhGG of fuel tank; 2 - shield; 3 - tank with auxiliary propellant components; 4 - ZhGG of oxidizer tank.

Key: (1). In the chamber/camara.

Fig. 13.16. Diagram DU with TGG: 1 - drain-reserve valve; 2 - TGG; 3 - unit of input/introduction of combustion products TGG into tank; 4 - membrane/diaphragm TGG; 5 - pressure indicator.

Key: (1). In the chamber/camera.

Page 222.

Usually solid fuel has an excess of fuel. Therefore solid-propellant gas generators it is expedient to use for the displacement of fuel. The displacement of oxidizer or monopropellant by the combustion products of charge of TGG is dangerous due to the possibility of their afterburning in the tank. Therefore for the target increases in the reliability of DU for the displacement of oxidizer use separate TGG with the solid-propellant grain, which has the excess of oxidizer, or provide for separating device.

However, in certain cases it is possible to allow/assume afterburning in the tank, if this does not decrease the reliability of the work of engine installation, since in this case is decreased the necessary flow of the displacing gas. In the extreme case the gas cushion/pad of tank can serve as peculiar liquid-gas generator, if we supply to the surface of liquid component fuel into the oxidizer tank and the oxidizer into the fuel tank.

Of all types of pressururized-propellant feed greatest structural/design simplicity possess the systems with the

solid-propellant gas generators. Furthermore, TGG provide the rapid output/yield of engine to the nominal rating, for which together with the basis is used starting/launching of TGG with the solid-propellant grain, which has high rate of combustion. The combustion products of the charge of starting/launching of TGG priming charge of basic gas generator and they rapidly increase pressure in the tanks. Additional advantage of TGG in comparison with the gas storage tank of pressure and ZhGG is the absence of gas containers high-pressure: since their complete hermetic sealing/pressurization/sealing, necessary for the rocket vehicles with prolonged period of storage or flight (especially under conditions of outer space), presents difficulties.

However, feed systems with TGG possess the number of shortcomings in comparison with the systems, which use a gas storage tank of pressure and ZhGG. To main disadvantages one should relate:

1. The complexity of the adjustment: it is in particular difficult to ensure stable (nonpulsating) burning in TGG (the shortcoming indicated possess systems with ZhGG).

2. Difficulties, which appear with power change and upon multiplying of engine.

3. Dependence of engine characteristic on ambient temperature,

which affects rate of combustion of solid fuel. For example, the engine thrust with an increase in the ambient temperature respectively increases/grows. The shortcoming indicated can be excluded, if on the upper bottom of tanks to place drain-reserve valves and to supply into the gas cushions/pads of tanks the greater expenditure of products of combustion how it is required for the creation of the prescribed/assigned pressure in the tanks. But excess expenditure of combustion products (it grows with an increase in the ambient temperature) bronzes through DPK into the environment. Therefore the mass of system with TGG proves to be more than the mass of system with ZhGG whose characteristics only to the low degree depend on ambient temperature.

The smallest mass have feed systems with ZhGG, but they are fairly complicated, especially during the utilization of two-component ZhGG, and they are used rarely.

Page 223.

§13.5. General/common/total characteristic of pump feed system.

In the pump feed system propellant component are included the following systems and the aggregates:

a) turbopump unit (Fig. 13.1);

b) the system, which creates certain excess pressure on entering the pumps;

c) starting system of turbine;

d) the power-supply system of turbine by gaseous working medium/propellant;

e) branch system of the exhaust (crushed) gas.

TNA is intended for increasing the pressure of the components of propellant and their supply into the chamber/camera and ZhGG. Gas turbine TNA develops power during the supplying to it of the gas flow, which possesses sufficiently high temperature and pressure. Pumps of TNA consume this power and are used it for increasing the pressure of the components of propellant and their supply into the chamber/camera and ZhGG.

Into the system, which creates overpressure on entering the pumps enter the aggregates of the supercharging/pressurization of the oxidizer tanks and fuel, and also the devices, which additionally raise the pressure of propellant component at the inlet into pump. Such devices include the jet and auger series-connected pumps which are considered in §13.10.

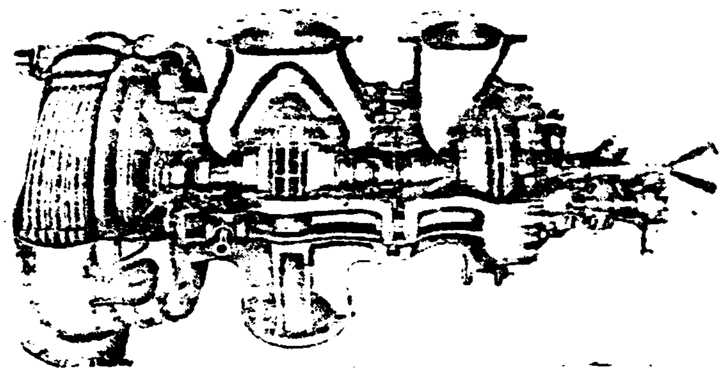

Fig. 13.17. TNA of ZhRD RD-107 "Vostok".

Page 224.

Starting system of turbine can encompass the cartridge starter (starting/launching TGG) or small tanks with the starting/launching propellant components; it works only in the period of the starting/launching of turbine, after which the latter changes to the feed/supply from the basic system (from ZhGG). Starting systems of turbine are examined in §14.1.

In the power-supply system of turbine by gaseous working medium/propellant are included ZhGG, the regulators and the valves, and also the conduits/manifolds, which supply propellant components to ZhGG.

Branch system of the crushed gaseous working medium/propellant of turbine is exhaust pipe for ZhRD with the throw-out of the working medium/propellant indicated into the environment and gas conductor for the engines, which work on the diagram "gas-liquid" and "gas-gas".

Exhaust pipe connect up to the exhaust collector turbines with the aid of the flanged or welded joint. Exhaust pipe is the thin-walled conduit/manifold, which is ended by divergent nozzle; its exit section they usually place at the level of nozzle exit section of the main engine chambers.

Gas conductor of ZhRD, which work on the diagram "gas-liquid" or "gas-gas" (latter/last type engines have two gas conductors), is the thick-walled conduit/manifold, which connects the exhaust collector of turbine with the head of chamber/camera.

§13.6. General-arrangement diagrams TNA.

Depending on the mutual arrangement/position of pumps and turbine are distinguished single-shaft and multishaft turbopump units (Fig. 13.18).

In single-shaft TNA the pumps of oxidizer and fuel, and turbine

also place on one shaft.

In twin-shaft TNA on one shaft are arranged/located the pumps of oxidizer and fuel, and on other - turbine or on one shaft they place the fuel pump and turbine, and on other - the pump of oxidizer.

In three-shaft TNA each pump and turbine assemble on the separate shaft.

Simplest construction/design has single-shaft TNA. However, it possesses the essential shortcoming: the number of revolutions of the shaft, on which are placed the turbine and pumps, is limited by one of the aggregates (usually by the pump of oxidizer), so that the number of revolutions of shaft proves to be optimal only for it, and other aggregates, in particular turbine, they work on the lowered/reduced relative to optimal number of revolutions.

Page 225.

Multishaft construction/design of TNA makes it possible for each aggregate to select the number of revolutions, which is optimal from the point of view its efficiency and low dimensions and mass; however, multishaft TNA has complex construction/design. The kinematic constraint between all shafts of TNA provides train of

reducing gears which works under the severe conditions (high peripheral of velocity, the large transmitted power). Gain in the mass of TNA, as a result of the optimal number of revolutions of turbine and each pump, can be brought to the minimum due to the large mass of reductor, systems of its lubrication and cooling.

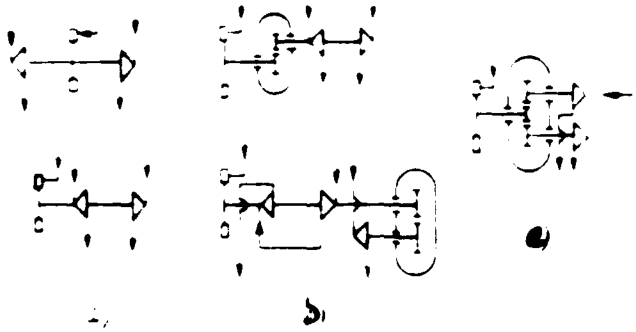

Fig. 13.18. The general-arrangement diagrams of single-shaft (a), twin-shaft (b) and three-shaft (c) TNA.

§13.7. Device of centrifugal pump.

For the pump feed of propellant components in ZhRD in essence use the centrifugal pumps, which possess low sizes/dimensions and mass and high efficiency.

In centrifugal pump (Fig. 13.19) are included housing, the impeller, shaft, bearings and packings/seals.

In pump casing it is possible to isolate intake pipe; the cavity in which is placed the impeller; volute/snail and diffuser. Through intake pipe the liquid is supplied to the intake part of the impeller. The torsion of the liquid before the impeller is eliminated by dividing wall, cast for one whole with intake pipe. With the

cantilever arrangement of the pump of the components of fuel/propellant it is supplied along the axis of shaft, in this case intake pipe has simplest form (usually the form of truncated cone with smallest area at the inlet into the impeller). However, from the designs frequently use the delivery of propellant components at angle of 90° to the axis/axle of pump spindle. In this case the form of intake pipe becomes complicated, especially if through it is passed pump spindle. Propellant component flows around about the shaft and also it is supplied to the intake part of the impeller all its over circumference.

The liquid, which escape/ensues at a high speed from the channels of impeller all over circumference, is assembled into the volute/snail and it moves in it towards diffuser.

Page 226.

The cross-sectional area of voluta/snail during the motion to the side of diffuser continuously increases/grows. Kinetic energy of liquid is converted into the potential pressure energy in the diffuser, so that great pressure a liquid has at the output/yield from the diffuser.

For high-pressure obtaining on output/yield of the pump instead

of the described above single-stage are used the two-stage pumps (Fig. 13.20), which are two series-connected single-stages pump, usually arranged/located on one and the same shaft.

Pump casing usually consists of strictly housing and cover/cap, decanted from the aluminum alloys and connect/joined together by flange joint. Pump casing, which creates high pressure, they cast made of steel.

Impellers most frequently are made by casting. Therefore all elements/cells of impeller are the unit forming its internal cavity, divided by blades into several identical channels. Usual impeller has not more than eight blades. Liquid enters the intake cavity of impeller, and then into the channels between the blades.

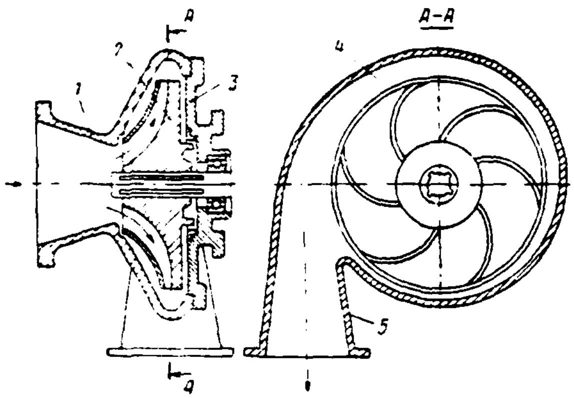

Fig. 13.19.

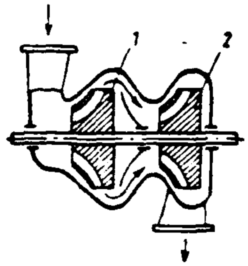

Fig. 13.20.

Fig. 13.19. Schematic of centrifugal pump: 1) intake pipe; 2) impeller; 3) cover/cap of housing; 4) volute/snail; 5) diffuser.

Fig. 13.20. Schematic of two-stage pump: 1) pump of first stage; 2) second pump.

Page 227.

The wall thickness of impeller exceeds 5 mm. To the shaft, on which on the splines is fit/mounted the impeller, is fed power. Therefore impeller is rotated and its blades exert pressure on the liquid, as a result of which the velocity and the pressure of liquid during the motion in the channels continuously increasing. From the impeller the liquid enters the volute/snail of housing. The impeller of the

described construction/design is called closed.

They frequently use, especially into ZhRD of low thrust, open impellers, whose cover/cap is absent. For them are characteristic the increased power losses to the overflowing of propellant component from high-pressure cavity (at the output/yield from the impeller) into low-pressure cavity (at the inlet into the impeller).

Distinguish impellers with the one-way (Fig. 13.21) and bilateral inlet liquids. Impellers with the bilateral inlet (Fig. 13.22) are two singleflow impellers connected by rear surfaces and decanted together. Such impellers make it possible to increase the permissible number of revolutions of pump spindle and due to this to lower mass TNA (see §13.9).

The efficiency of pump to a considerable degree depends on fluid flow rate, which flows from high-pressure cavity in low-pressure cavity. For reducing/descending the expenditure indicated are used the slot, labyrinth and floating packings/seals of impellers.

Slit gaskets (Fig. 13.23a) decrease the overflowing of liquid as a result of low radial clearance δ between the graphite ring, securely fastened in the groove of pump casing, and the cylindrical annular groove of impeller.

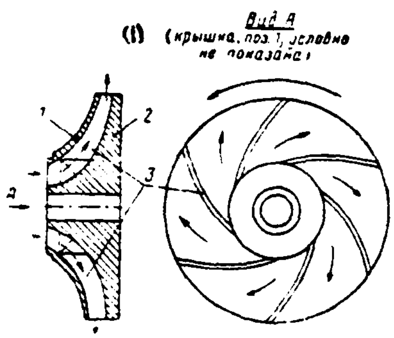

Fig. 13.21.

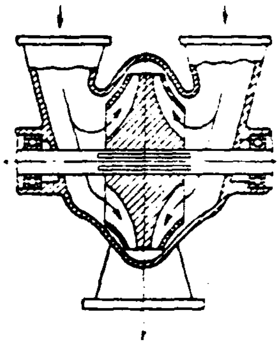

Fig. 13.22.

Fig. 13.21. Singleflow impeller 1) cover/cap: 2) housing; 3) blade.

Key: (1) . View A (cover/cap, pcs. 1, are not conditionally shown).

Fig. 13.22. Centrifugal pump with impeller, which has bilateral inlet.

Page 228.

Losses to the overflowing during such packings/seals compose approximately/exemplarily 15o/o of the general/common/total expenditure through the impeller.

Labyrinth seals (Fig. 13.23b) are more effective than slot, since labyrinth grooves impede the overflowing of liquid.

The floating packings/seals (Fig. 13.23c), which consist of the set of the alternating disks from a fluorine-containing polymer or aluminum, make it possible to lower the overflowing of liquid to the insignificant value (not more than 5o/o of the general/common/total expenditure through the impeller).

So that the liquid, which flows through the packings/seals, would not disturb the course of the main flow of liquid at the inlet into the impeller, in pump casing is provided for the special deflector, which is guided the flowing liquid in the direction of the main flow (Fig. 13.23b and c).

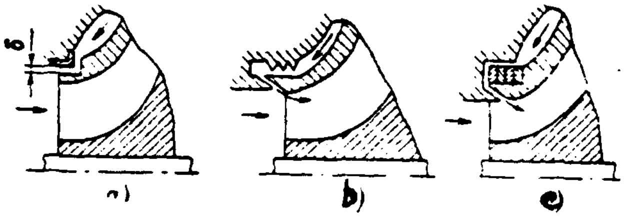

Fig. 13.23. Packings/seals of the impeller: a) slot; b) labyrinth; c) floating.

§13.8. Basic parameters of pump.

For calculating the pump basic are the following parameters.

1. Volumetric fluid flow rate through pump $\dot{v}$. It is determined according to the mass propellant component flow $\dot{m}$, found from the thermal design, according to the formula

$$\dot{v}=\frac{\dot{m}}{\varrho},$$

where ρ - density of propellant component at an inlet temperature into the pump.

The rate of the action of liquid in the pump, and therefore, also the volumetric flow rate $\dot{v}$ of the directly proportional to number of revolutions of its impeller:

$$\dot{v}\sim n.$$

Therefore for changing the fluid flow rate through the pump it is

necessary respectively to change the number of revolutions of impeller.

Page 229.

2. Pressure H. The pressure, created by pump, is an increase in energy of one kilogram of the liquid, passing through the pump. If we consider that the velocity of liquid at the inlet into the pump is equal to outlet velocity from it, then pressure is expressed by the formula

$$H=\frac{p_{вых}-p_{вх}}{\varrho g_з}, \quad (13.1)$$

where $p_{вх}$ and $p_{вых}$ the pressure of liquid at the inlet into the pump and at the output/yield from it respectively;

$g_з$ - acceleration of gravity at the level of sea, equal to 9.80665 m/s².

Pressure H is expressed in the meters of liquid column, and the pressure increase, provided by pump, depends on pressure and density of the liquid:

$$\Delta p_{нас}=p_{вых}-p_{вх}=H\varrho g_з.$$

Pressure is directly proportional to the square of number of revolutions n and to the square or outside the diameter of impeller $D_{кр}$:

$$H\sim n^2D^2_{кр}.$$

For an increase in the pressure it is necessary to raise the number of revolutions of impeller or to apply impeller with the large outside diameter.

Necessary pressure for output/yield from the pump of ZhRD can be designed from the formula

$$p_{\text{вых}} = p_{\text{к}} + \Delta p_{\text{ф}} - \Delta p_{\text{м}},$$

where $\Delta p_{\text{м}}$ - all types of hydraulic losses on the main of the corresponding propellant component from the pump to the injectors of chamber/camera.

3. Number of revolutions of shaft n. If we designate the number of revolutions of shaft per minute n, and angular velocity of shaft - ω, then the relationship/ratio between them takes the following form:

$$\omega = \frac{2\pi n}{60}.$$

4. Useful lifting power $N_{\text{ж}}$. Useful is the power, transmitted by the pump of liquid, i.e., the power, spent on the creation of real pressure with that determined volumetric flow rate; it is determined from the formula

$$N_{\text{ж}} = \dot{v} H \varrho g,$$

or taking into account equation (13.1)

$$N_{\text{ж}} = \dot{v}(p_{\text{вых}} - p_{\text{вх}}). \qquad (13.2)$$

Page 230.

during the recording in the form of equation (13.2) net power does not depend on the density of liquid ρ.

5. Efficiency of pump $\eta_{нас}$. The efficiency of pump considers all types of losses in it and it can be written in the following form:

$$\eta_{нас} = \eta_{об}\eta_{гид}\eta_{мех},$$

where $\eta_{об}$, $\eta_{гид}$ and $\eta_{мех}$ - volumetric, hydraulic and mechanical efficiency respectively.

Volumetric efficiency $\eta_{об}$ characterizes the losses, connected:

a) with the overflowing of liquid from the cavity at the output/yield from the impeller into the cavity at the inlet into it;

b) with the leakage of liquid into the drainage cavity, from which it is abstracted/removed outside.

Due to the presence of the losses indicated (let us designate their $\Delta\dot{v}_{ут}$) fluid flow rate through impeller $\dot{v}_{кр}$ it is more than the expenditure through the pump:

$$\dot{v}_{кр} = \dot{v} + \Delta\dot{v}_{ут}.$$

The volumetric efficiency of pump is designed from the equation

$$\eta_{o6} = \frac{v}{v - \Delta v_{yT}}.$$

for the large/coarse pumps of ZhRD η_{o6} = 0.90-0.95.

Hydraulic efficiency η_{rmx} is the characteristic of hydraulic losses in the pump which include:

a) loss to fluid friction against the walls of channel and losses, connected with internal fluid friction as a result of its viscosity/ductility/toughness; let us designate the losses indicated Δh_{Tp};

b) the losses, connected with the nonconformity of the directions of the fluid flow and channels in the pump: loss by shock and flow separation at the inlet into the impeller, the diffuser, the volute/snail and the exhaust duct; these losses let us designate $\Delta h_{y\pi}$.

Losses to the friction are proportional to the square of expenditure or velocity:

$$\Delta h_{\text{тр}} \sim \dot{v}^2 \ (\overset{(1)}{\text{или}}\ W_{ж}^2)$$

Key: (1). or.

and are removed/taken during the reduction in area of internal canal surface of pump and an improvement in the purity/finish of their processing/treatment.

Losses $\Delta h_{\text{уд}}$ have the small value with the work of pump in the nominal rating, i.e., when $\dot{V}=\dot{V}_{\text{ном}}$. when $\dot{V}>\dot{V}_{\text{ном}}$ and $\dot{V}<\dot{V}_{\text{ном}}$, with the deviation of the mode/conditions of the work of pump from the nominal, loss $\Delta h_{\text{уд}}$ increase/grow.

Hydraulic losses lead to certain increase in the temperature of liquid with its passage through the pump. For the pumps of ZhRD $\eta_{\text{гид}}=$ 0.7-0.9.

Page 231.

Mechanical efficiency $\eta_{\text{мех}}$ characterizes power losses to bearing friction and packings/seals, and also to the friction of the nonoperative (not creating pressure) surfaces of impeller against the liquid. For the pumps of ZhRD $\eta_{\text{мех}}=0.85-0.98$, the high values $\eta_{\text{мех}}$ corresponding to the pumps of greater power (large sizes/dimensions).

The overall efficiency of pumps of ZhRD $\eta_{\text{нас}}=0.5-0.85$.

For supplying the volumetric fluid flow rate $\dot{V}$ with the density

ρ under pressure H it is necessary to the pump spindle to supply the power, designed from the equation

$$N_{нас} = \frac{N_г}{\eta_{нас}} = \frac{\dot{v}H\rho g_3}{\eta_{нас}} = \frac{\dot{v}(p_{вых} - p_{вх})}{\eta_{нас}}.$$

Efficiency $\eta_{нас}$ it is possible to determine according to the results of testing (spill) the pump, for which it is necessary to measure values of $\dot{v}$, $p_{вх}$, $p_{вых}$ and n, and also the shaft torque $M_{кр}$.

In terms of known values $M_{кр}$ and n it is possible to determine consumed lifting power:

$$N_{нас} = \frac{2\pi n}{60} M_{кр} = \omega M_{кр}.$$

Since $\dot{v} \sim n$ and $H \sim n^2$, then $N_{нас} \sim n^3$.

§13.9. Selection of the number of revolutions of shaft TNA cavitation.

The important stage of design of TNA and engine installation with ZhRD as a whole is the selection of the optimal number of revolutions of shaft of TNA.

The need for the selection of the optimal number of revolutions of single-shaft TNA can be explained by the following. With an increase in the number of revolutions increases/grows the efficiency of turbine with its invariable sizes/dimensions or is decreased the

necessary outside diameter of turbine (and, consequently, its mass) with the invariable efficiency. Furthermore, is decreased the necessary outside diameter of pump and its mass. But with an increase in the number of revolutions increases/grows the velocity of liquid at the inlet into the impeller, which can lead to the inadmissible mode/conditions of the work of pump. In this case it is necessary to increase the pressure of liquid at the inlet into the pump. If the pressure indicated is increased by increasing the boost pressure in the tanks, then appears the need for thickening their walls (with the appropriate increase of their mass).

Is not admitted the work of pump in the mode/conditions of the so-called cavitation. Cavitation (from the Latin word cavitas - vacuum) is the process of forming bubbles of steam in those zones of liquids in which the static pressure is less than the pressure of the saturated steams (i.e., the temperature of liquid exceeds the temperature of its boiling), and also the process of the subsequent filling of the bubbles indicated with liquid during their incidence/impingement into the zone of elevated pressure.

Page 232.

The pressure of liquid small in the area of the flow around the intake edges of blade of impeller; therefore cavitation appears first

of all in the cross section at the inlet into the impeller.

Are distinguished the modes/conditions of partial and complete cavitation.

During the mode/conditions of partial cavitation the formed bubbles of steam manage to be filled in the intake part of the impeller; they are filled at a very high speed, which leads to the hydraulic impacts on the internal canal surface of impeller and to their erosion. With the continuous operation in the mode/conditions of partial cavitation the erosion of canal surface of impeller considerably decreases its strength and it is possible to lead to the destruction vane. But the operating time of pumps of ZhRD is relatively small, and therefore the mode/conditions of partial cavitation is permitted. Here it is possible to attain the perceptible gain in the mass of tanks due to decompression of their supercharging/pressurization.

If bubbles of vapor do not manage to be filled up with liquid in the impeller (i.e., through the impeller goes not liquid, but the mixture of liquid and vapor), then begins the mode/conditions of complete cavitation, which is accompanied by a sharp loss of pressure and fluid flow rate through the pump; this mode/conditions of the work of pumps ZhRD inadmissible.

A change in the pump head with decompression at the inlet into the impeller is estimated by separation cavitation characteristic (Fig. 13.24). It build for the cross section at the inlet into the impeller on the basis the spills of pump at constant values of n and $\dot{m}$, moreover first provide great pressure liquids at the inlet into the pump, and then with each subsequent spill the pressure gradually decrease and they finish to the value, at which a loss of pressure composes 2-3o/o. This pressure designate $p_{вх.кав}$.

Working conditions for noncavitation can be written in the form

$$p_{вх} > p_{вх.кав}.$$

Optimal number of revolutions at the prescribed/assigned pressure at the inlet into the pump is the maximum permissible number of revolutions $n_{max\,доп}$. It are designed from the formula which establishes the bond between values $H_{вх}, H_s$, $\dot{V}$ and n, moreover

$$H_{вх}=\frac{p_{вх}}{\varrho g_з} \overset{(1)}{и} H_s=\frac{p_s}{\varrho g_з}.$$

Key: (1). and.

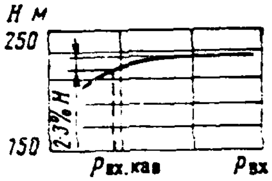

Fig. 13.24. Separation cavitation characteristic of centrifugal pump.

Page 233.

The formula indicated takes the following form:

$$n_{\text{вх доп}}=\frac{C}{\sqrt{v}}\left(\frac{H_{\text{вх}}-H_s}{10}\right)^{3/4}, \qquad (13.3)$$

where C - constant for this pump value (cavity coefficient, or the coefficient of Rudnev). The higher the coefficient of C, the better the anticavitation qualities of pump.

Coefficient C depends on the design features of impeller and is determined experimentally. For the usual centrifugal pumps C=800-1100, and during the utilization of blades of special shape in the impeller it is increased to 2200.

The effective method of increasing the anticavitation properties of centrifugal pump is the installation of the low-pressure jet or auger series-connected pump at its inlet. For example, during the

installation of helical-type pump with the sufficiently large length of blades the cavity coefficient C increases/grows to 3500-4000 [30].

From equation (13.3) is evident that to increase the number of revolutions of pump (and to respectively decrease its sizes/dimensions) is possible (besides an increase in the cavity coefficient of C) in the following cases:

a) with an increase in the pressure at the inlet into pump $p_{вх}$ (pressure $H_{вх}$);

b) with the decrease of the pressure of the saturated vapors of liquid p_s (pressure H_s) and volumetric fluid flow rate through the impeller.

Pressure $p_{вх}$ can be increased by increasing the boost pressure of tanks; however, this way is least advantageous due to the need for an increase in their wall thicknesses.

The pressure of the saturated vapors of liquid p_s depends on type and temperature of liquid. Due to high values p_s for the cryogenic propellant components it is difficult to ensure the noncavitation mode/conditions of the work of pumps at low pressures $p_{вх}$.

The volumetric flow rate through the impeller can be decreased by the utilization of impellers with the bilateral inlet (see Fig. 13.22): fluid flow rate through each impeller is decreased 2 times and in accordance with formula (13.3) it is possible in $\sqrt{2} \approx 1.4$ the time to raise the number of revolutions of pump, after leaving invariable pressure p_{BX}. Since volumetric oxidizer consumption is usually more than the volumetric flow rate of fuel, then during the single-shaft construction/design of TNA the maximum permissible number of revolutions is limited by the pump of oxidizer and impeller with the bilateral inlet it is expedient to use first of all for it.

§13.10. Systems of pressure increase at the entry into the pump.

The necessary pressure at the inlet into the pump is created with the aid of the pressurized system of tanks and special series-connected pumps.

Page 234.

Pressurized systems of tanks. Tank pressurization consists of the maintenance of the corresponding overpressure in their gas cushions/pads by supplying in them the gas. Therefore the pressurized

systems of tanks in many respects are analogous to the pressure feed systems of propellant components with one difference: the boost pressure of tanks usually substantially less than pressure in the tanks during close supply is 2-4 bars [≈2-4 kgf/cm²] and in certain cases it only reaches 6 bars [≈6 kgf/cm²].

Are distinguished two types of tank pressurization: by cold and hot gas. Tank pressurization by hot gas (for example, with the aid of special ZhGG) causes certain complication of engine installation, but provides a noticeable reduction/descent in its mass and therefore it is extensively used, especially in the large/coarse carrier rockets.

Liquid-oxygen tanks, liquid hydrogen and nitrogen tetroxide can be forced via selection at the output/yield from pump and vaporization of their small portion in the heat exchanger which usually is arranged/located on the line of exhaust gas, i.e., after turbine, including in its output collector/receptacle (see Fig. 13.17).

Supercharging/pressurization by cold gas is accomplished/realized by a system with the gas storage tank of pressure. For the initial section of missile trajectory, passing through the sufficiently dense layers of the atmosphere, it is possible to use cold tank pressurization, using the incident airflow.

Installation of the preconnected pump at the inlet into the pump. Are distinguished the jet and auger series-connected pumps.

The jet series-connected pump (or ejector) by construction/design and according to operating principle is analogous to the jet pump, examined in the beginning of present chapter, but for the work of the jet series-connected pump is used not gas, but the small part of the propellant component flow, selected/taken at the output/yield from the basic pump.

The jet series-connected pumps are installed in the coarse-wire/coarse-conductor, which supplies propellant component to the pump. The use/application of the jet series-connected pumps is sufficiently effective, with what their low efficiency to a considerable degree is purchased by low mass and simplicity of construction/design.

Auger (or booster) series-connected pump is the variety of axial-flow pump and are most frequently two or three spiral blades of trapezoidal form with the constant space and with the angle of ascent of 3-7°. The auger series-connected pump possesses substantially higher anticavitation properties, than a centrifugal pump. This is

explained by the following factors.

1. Pressure difference between working (creating pressure) and nonoperative sides of blade for axial-flow pump is less than for centrifugal, i.e., load on blade of axial-flow pump is less than to blade of centrifugal pump.

Page 235.

2. In centrifugal pump liquid moves in the direction of effect/action of centrifugal forces which break away liquid from steam bubbles, which contributes to course of cavitation.

In the jet pump the liquid moves in essence over its axis/axle. If vane channels have sufficiently large length, then the cavitation, which arose in the initial part of helical-type pump, can not have an effect on its work, since bubbles of vapor that are generated on the peripheral part of the blade, are compressed by the liquid which is ejected by centrifugal forces from blade root to its periphery.

The pressure, created by the auger series-connected pump, composes 3-20o/o of the general/common/total pressure and is limited to the fact that the efficiency of the pump indicated are less than the efficiency of centrifugal pump, and also the fact that with an

increase in the pressure, created by the auger series-connected pump, i.e., with an increase in the load on the blade, deteriorate its anticavitation properties.

The outside diameter of worm screw is determined by the diameter of the intake pipe junction of centrifugal pump. The inner diameter of worm screw must be smallest, since with an increase in the surface of blades is decreased load on them and are improved the anticavitation properties of the auger series-connected pump.

The number of revolutions of the series-connected pump and the centrifugal pump can be one and the same or different. In the first case is provided large simplicity of construction/design of TNA. However, decreasing the number of revolutions of the series-connected pump relative to the number of revolutions of centrifugal pump, it is possible to decrease the load on the blades of worm screw and thereby to additionally raise its anticavitation properties; this series-connected pump needs special drive. It is most expedient to use for this purpose of hydroturbine, which work on the heated component of propellant (for example, on the propellant component, selected/taken from the coolant passage of chamber/camera) [1].

The auger series-connected pumps, especially with the hydroturbines, increase mass of TNA and complicate its

construction/design, but their use gives the perceptible effect in connection with the possibility of decompression in the tank pressurization.

§13.11. Construction of turbine.

In turbine (Fig. 13.25) are included:

a) the housing, which consists of the cover/cap in which is placed nozzle cascade, and exhaust collector;

b) blade wheel;

c) the guides of blade (for the double-staged turbine).

d) shaft with the clutch.

Page 236.

Gas enters the collector/receptacle of nozzle cascade, and from it - into the nozzles, in which it is accelerated/dispersed to the large velocity (1000 m/s and more) and it falls to the rotor blades. If turbine is two-stage, then between the rotor blades of first and second stages place fixed guides of the blades which turn gas in

order to bring it to the rotor blades of the second step/stage at the required angle.

Pressure and the temperature of crushed gas (at the output/yield of the rotor blades) is substantially less than at the inlet into the turbine. Crushed gas enters exhaust collector, and from it - into exhaust pipe or gas conductor.

Nozzle cascade is the weldment, which distributes the applied gas in the circumference of turbine and feeding it with the high velocity to the rotor blades. If gas is supplied to the turbine rotor blades all over circumference (degree of the admission of turbine $\varepsilon=1$), then the nozzles of nozzle cascade place evenly all over circumference of turbine. If the degree of admission is lower than one ($\varepsilon<1$), then nozzle assembly is a ring segment whose length in the circumference is proportional to the degree of admission. The axis of the symmetry of nozzles is located at medium altitude of rotor blades. Nozzle cascade is the most heat-stressed unit of turbine.

Turbine disk has the thickened hub (for the fastening to the shaft) and the thickened hoop (for the installation of rotor blades), which are connected between themselves by the shaped part of the disk.

Turbine rotor blades consist of foot and shaped part (foil), made for one whole.

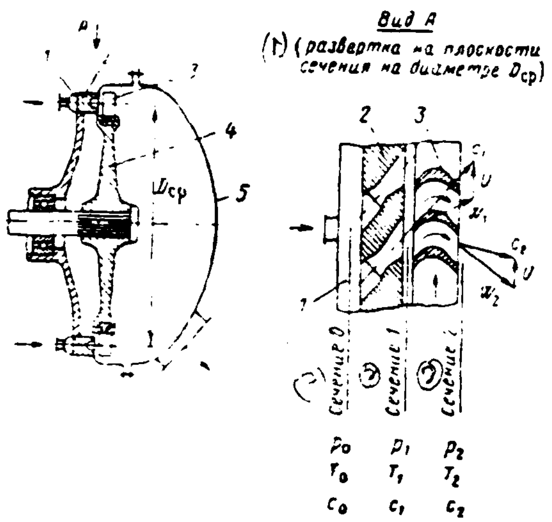

Fig. 13.25. The schematic of the single-stage axial-flow turbine: 1 - inlet manifold; 2 - nozzle cascade; 3 - rotor blades; 4 - disk; 4 - exhaust collector/receptacle.

Key: (1). View A. development/scan on plane of cross-section at the diameter). (2). cross section.

Page 237.

Foot fastens blade to the hoop of disk; are distinguished inarticulate and scarf joints of blades with the hoop of disk.

Inarticulate are welded and soldered joints. To cup is used welded joint (Fig. 13.26a).

From the locking ones are most widely used the connections with the T-shaped lock (see Fig. 13.26b).

The guides of the blade of double-staged turbines are fastened to the segments which are connected with turbine casing by flange (Fig. 13.27).

R The exhaust collector of turbine with its cantilever arrangement on the shaft is the welded housing which is fastened to turbine casing with the aid of the flanged or welded joint.

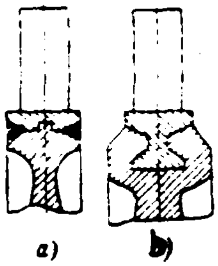

Fig. 13.26.

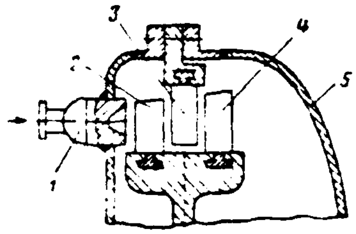

Fig. 13.27.

Fig. 13.26. Types of connections of blades with turbine disk: a) welded; b) with T-shaped lock.

Fig. 13.27. Schematic of flow area of two-stage impulse turbine: 1 - nozzle cascade; 2 - rotor blades of the first stage; 3 - guide blades; 4 - rotor blades of second step/stage; 5 - exhaust collector.

§13.12. Classification and the operating principle of turbines.

Turbine must possess low dimensions and with a mass of; the gas flow for the creation of the prescribed/assigned shaft horsepower of TNA, it must be also lowest possible.

For the examination of the principle of the work of turbine it is expedient to isolate the following cross sections of its flow area (see Fig. 13.25).

1. Cross section 0 - in supplying collector/receptacle of nozzle cascade at inlet into its nozzles.

2. Cross section 1 - at nozzle outlet of nozzle cascade (at inlet into rotor blades).

3. Cross section 2 - at output/yield from rotor blades, i.e., in exhaust collector.

In the cross section 0 gas possess the greatest enthalpy. With the flow of gas along the nozzles occurs its expansion during which the part of the enthalpy is converted into the kinetic energy, in this case the pressure and the temperature of gas are decreased, and its velocity increases/grows.

Page 238.

Gas, flowing out behind the nozzles with large speed, enters the rotor blades and during the action in the channels between them is given up the large part of its energy to rotor blades, which leads to the origination of circumferential force to the blades and creation of shaft horsepower of turbine.

On the special features/peculiarities of expansion gas during its motion along the flow part distinguish active and reaction turbines.

In the impulse turbines the gas is expanded only during the motion along the nozzles of nozzle cascade, and the flow of gas between the rotor blades occurs at a constant pressure.

In the reaction turbines the gas is expanded both in the nozzles of nozzle cascade and during the motion between the rotor blades.

Fig. 13.28 depicts the section/cut of flow area and the graphs of a change in the parameters at the flow of gas along flow area of the active and reaction turbines.

In the impulse turbine the gas enthalpy during the motion along the nozzles of nozzle cascade is decreased, and during the motion between the blades in the absence of losses it remains constant. However, due to the losses to the friction and the vortex formation enthalpy during the motion between the blades somewhat increases/grows, and relative gas velocity (gas velocity relative to blades) is decreased.

In the reaction turbine the gas enthalpy is decreased during the motion both in the nozzles of nozzle cascade and during the motion between the rotor blades, i.e., gas enthalpy is decreased in the reaction turbine just as its pressure.

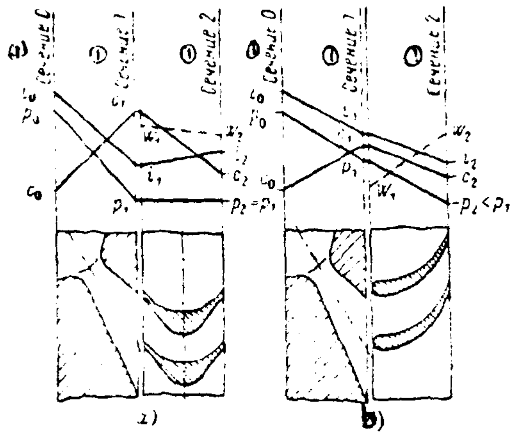

Fig. 13.28. Diagram of flow area and the graphs of a change in the parameters of gas in its length for the active (a) and reactive/reagent (b) turbines.

Key: (1). cross section.

Page 239.

§ 13.13. Basic parameters of turbine.

A number of bases includes the following parameters of turbine.

1. Available power of turbine, i.e., power of turbine on shaft; it must be equal to sum of powers, consumed by pumps of oxidizer and fuel, and also by pumps of auxiliary components of propellant (in presence of latter), i.e.,

$$N_{\text{турб}} = N_{\text{нас.ок}} + N_{\text{нас.г}} + N_{\text{нас.всп}}$$

and is determined from formula

$$N_{\text{турб}} = \eta_{э} L_{\text{ад}} \dot{m},$$

where $\eta_{э}$ — overall efficiency of turbine (see § 13.14);

$\dot{m}$ — gas flow rate per second, which enters from turbine;

$L_{\text{ад}}$ — adiabatic work of expansion 1 kg. of gas, designed from formula

$$L_{\text{ад}} = \frac{k}{k-1} R T_0 \left[1 - \left(\frac{p_2}{p_0}\right)^{(k-1)/k}\right].$$

2. Pressure differential on turbine (expansion ratio of gas in turbine), equal to ratio p_0/p_2. Are distinguished high-gradient (p_0/p_2=15-40) and low-pressure (p_0/p_2=1.3-1.8) turbines. Pressure p_2 is called counterpressure.

The high-gradient ones include the turbines with the throw-out of crushed gas into the environment; in the nozzles of their nozzle cascade is operated/worn supercritical gas pressure drop. For increasing the power of turbine it is desirable to ensure the greater expansion of gas; at the invariable pressure p_0 of this it is possible to attain, decreasing pressure p_2. But so that to the mode/conditions of the work of turbine and, consequently, also TNA as a whole would not affect a change in the ambient pressure, pressure p_2 is necessary to select more than maximum ambient pressure: $p_2 \approx 1{,}3 p_{h\,max}$ (taking into account to the possibility of the work of Laval nozzle turbine exhaust under the conditions of overexpansion [17]). In this case on the nozzle of turbine exhaust is provided a supercritical pressure differential, as a result of which the nozzle, as it was noted into § 9.1, develops certain thrust. Specific impulse $I_{уд}$ of the nozzle of exhaust pipe is less than the value $I_{уд}$ of chamber/camera, and with the increase of the gas flow through the turbine value $I_{уд}$ of engine descends. Therefore the prescribed/assigned power of high-gradient turbines it is expedient to obtain with the lowest possible gas flow through them.

Page 240.

For the turbines ZhRD, which work on the diagram the "gas - liquid" or "gas- gas", is characteristic the high gas flow: for example, for the diagram " gas- liquid" it is usually equal to complete flow rate of one of the components of propellant and certain part of the flow rate of another. Therefore for such ZhRD are used the turbines, which develop sufficient power with a subcritical pressure differential, i.e., low-pressure turbines.

3. Turbine inlet gas temperature T_0. Temperature T_0 together with the expansion ratio of gas determines the adiabatic work of expansion 1 kg. of gas, which increases with its increase. Depending on material of blades and duration of the work of engine temperature T_0 is selected in limits of 750-1200°K.

4. Number of revolutions of shaft of turbine n. Number of revolutions n during the single-shaft construction/design of TNA is determined from the condition for the noncavitation work of pumps, and during the multishaft construction/design - from the condition the greatest efficiency of turbine and its smallest dimensions.

In calculations the turbines use peripheral speed U - the speed of the point, arranged/located on medium altitude of blade (at diameter D_{cp}), in this case

$$U=\frac{\pi D_{cp} n}{60}\ м/сек.^{(1)}$$

Key: (1). m/s.

§ 13.14. Efficiencies. Turbines and selection of relation U/c_1

With the work of turbine occur the losses:

a) in the nozzles of nozzle cascade;

b) on the rotor blades;

c) with outlet velocity;

d) to the friction of disk against the gas and ventilation;

e) mechanical.

All forms of the losses indicated considers the overall efficiency of turbine.

Losses in the nozzles of nozzle cascade and on the rotor blades

depend on the degree of the perfection of flow area of the turbine, including of the finish of the surface of nozzles and rotor blades and of their profile/airfoil.

Losses with outlet velocity are explained by the fact that gas at the output/yield from the rotor blades possesses certain speed c_2, i.e., kinetic energy of gas is used in the turbine not completely.

At the prescribed/assigned values of the gas velocity c_1 and angle of the slope of velocity vector c_1 to the plane of turbine disk α_1 minimum speed c_2 and, consequently, also smallest losses with outlet velocity reach at ratio U/c_1, which is determined from the formula

$$\frac{U}{c_1}=\frac{\cos\alpha_1}{2}. \qquad (13.4)$$

Page 241.

Usually in the impulse turbines TNA of α_1=15-20° and the speed c_1=1000-1400 m/s, in this case the necessary peripheral speed U, designed from formula (13.4), is obtained inadmissibly to large; in particular, sharply they are increased dimensions and the mass of turbine. Therefore in the high-gradient turbines frequently peripheral speed U is selected in the limits of 250-350 m/s, and

ratio U/c_1=0.1-0.3, which conditions losses with outlet velocity.

At the low values of ratios U/c_1, which it is expedient to use in the turbines TNA, efficiency of double-staged turbine substantially higher than in single-stage.

If losses to the friction of disk against the gas are inherent in each turbine, then windage losses - only partial, they increasing/growing in proportion to the decrease of the partiality of turbine.

The overall efficiency of high-gradient turbines is within the limits of 0.3-0.7, and low-gradient turbines for which U/c_1=0.4-0.6, has higher values.

In essence in ZhRD are used the axial-flows turbine, in which the gas moves in parallel to the axis/axle of shaft.

Specific to construction/design the so-called radial-flow turbines in which the gas moves over the radius of disk to the axis/axle of shaft (inward-flow turbine) or from the axis/axle of shaft to the periphery of disk (centrifugal turbine). Of the radial-flow turbines greater use/application found low-pressure inward-flow turbines.

§ 13.15. The liquid-gas generators.

The liquid-gas generator of the power-supply system of turbine of TNA produces the gas, which possesses sufficiently high ones by pressure and by temperature.

In ZhRD are used mono- and two-component ZhGG, which, as it was shown into § 9.1, can work both on the bases and on the additional propellant components.

Most extensively are used the two-component gas generators, which work on the basic propellant components. In ZhRD with selection of the exhaust gas from the turbine into the environment for the work of two-component ZhGG at the output/yield from the pumps is selected/taken small part (usually 2-3o/o) of the general/common/total consumption of basic propellant components.

The temperature of generator gas does not usually exceed 1200°K. If we supply to the turbine gas with the greater temperature, then the strength of material of blades noticeably descends or occurs the fusing of the blades and other elements/cells on the main of generator gas. The required temperature of gas of two-component ZhGG

is provided with the significant excess of oxilizer or fuel (see § 9.1).

Are distinguished also one-region and two-region ZhGG (Fig. 13.29).

Page 242.

In one-region ZhGG entire propellant component flow is supplied from the side of head, i.e., just as into the basic chamber/camera ZhRD.

In two-ozone ZhGG the part of excess propellant component is introduced inside ZhGG through the additional belt/zone of injectors, situated in the middle part of the generator. In such ZhGG it is possible to isolate two zones: the high-temperature (2000-3500°K) zone (from the head to the cross section, in which it is arranged/located the additional belt/zone of injectors) and zone with substantially the lower temperature (from the belt/zone indicated to the output/yield from ZhGG).

Two-zone ZhGG structurally/constructionally more complex than one-region ones and are used when in the one-region ones does not succeed in ensuring the stable process of burning or their length is

large due to the insufficiently active process of burning, caused by excess of one of the propellant components.

For equalization of temperature field at the output/yield from ZhGG, which has high value for the exception/elimination of fusings on the vain of generator gas, reduced length for ZhGG is taken more than for the combustion chamber.

Usually ZhGG have external flowing cooling, which provides their reliability and prolonged resource/lifetime of work; at a relatively low temperature of generator gas the necessity for this cooling drops off.

One-component ZhGG. In a number of cases it proves to be more worthwhile to use not two-component, but one-component ZhGG, in which in the presence of catalyst is accomplished/realized the decomposition of one-component liquid propellant (for example, peroxide of hydrogen) with the liberation of heat and the formation of gaseous products; this decomposition is called catalytic.

Can be used both solid and liquid catalysts, moreover the latter must continuously be supplied in ZhGG (this generator is actually two-component). Solid catalyst they place directly in ZhGG in the form of bundle (Fig. 13.30). Generators with the solid catalyst simpler by the construction/design are used more widely.

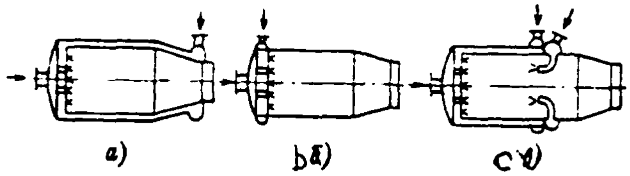

Fig. 13.29. Two-component ZhGG: a) the cooled one-region; b) the uncooled one-region; c) the cooled two-zone.

Page 243.

The bundle of solid catalyst for the decomposition of peroxide of hydrogen is grains of the solid basis-carrier (gypsum, cement, etc.), impregnated with the catalytically active salts (for example, $KMnO_4$), or pressed grids from the reactive metal (nickel, monel metal, brasses, etc.).

As catalyst for the decomposition of hydrazine can serve grids from the metals of platinum group.

The temperature of the generatrices of the decomposition products of peroxide of hydrogen (mixture of water vapor and gaseous oxygen) increases/grows with an increase in peroxide concentration of hydrogen and with the 80-90o/o concentration composes 720-1030°K. The

temperature of the decomposition products of hydrazine can be obtained within the limits from 875° to 1475°K by changing the retention time of hydrazine in the catalyst bed and change in the length ZhGG (by control of the degree of the decomposition of hydrazine).

During sizing of the bundle of solid catalyst are used the following specific parameters.

1. Specific catalyst surface area - area of surface area of catalyst, which falls per unit volume. For the series/number of the catalysts used the specific surface area composes 8-80 cm^2/cm^3.

2. Specific load of catalyst - maximally allowable consumption of liquid propellant component, which falls on 1 kg. of catalyst,

$$s=\frac{\dot{m}_{ж.к}}{m_{кат}}\frac{кг'сек}{(1)кг}.$$

Key: (1). kg/s/kg.

For example, for the solid catalyst, which consists of permanganate of calcium $CaMnO_4$ and chromate potassium, during the utilization of 80o/o peroxide of hydrogen value s composes 2.5-2.6 kg/s/kg.

With an increase in specific surface area and specific load of

catalyst is decreased the necessary volume of catalyst bed, and consequently, volume and the mass of generator.

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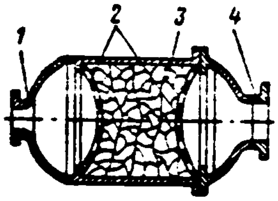


Fig. 13.30. One-component ZhGG: 1 - intake pipe; 2 - grid for the retention of solid catalyst; 3 - bundle of solid catalyst; 4 - exhaust duct.

Page 244.

Chapter XIV.

Starting systems, change in the mode/conditions and disconnection ZhRD. Systems of the creation of control forces and moments/torques.

§ 14.1. Starting systems ZhRD.

Starting system ZhRD must provide sufficiently rapid, but soft (without large fluctuations of pressure p_K) and reliable output/yield of engine on the nominal rating of work with the low unproductive expenditures of fuel/propellant.

The conditions of reliable starting/launching of ZhRD they are:

a) the absence of overshoot of pressure p_K over the permissible value (it can be caused by the accumulation of a large quantity of propellant components in the chamber/camera before their inflammation); furthermore, must be excluded the formation of explosive compound in the chamber/camera;

b) the low level of the pulsation of the pressure of combustion

products in the chamber/camera and the gas generator;

c) the small deviation of coefficient × in the chamber/camera and ZhGG from computed values.

Engine starting is the most complex and critical period of its work. A great quantity of emergencies of engines occurs precisely during this period. The parameters in the chamber/camera and the gas generator at engine starting continuously change, and engine passes through many modes/conditions, check and the study of each of which it is virtually impossible. Therefore the adjustment of starting/launching usually causes the great difficulties which increase/grow with an increase in the sizes/dimensions of chamber/camera.

Methods of launching of ZhRD. Are distinguished two methods of launching of ZhRD: stepless (smooth or "gun") and stepped (Fig. 14.1).

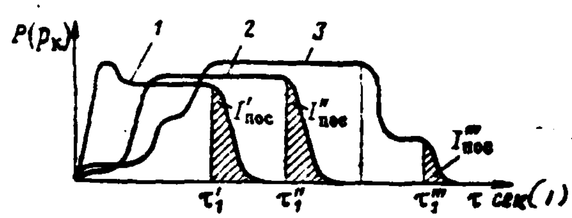


Fig. 14.1. A change in the thrust (pressure pк) with different types of starting/launching and disconnection of ZhRD: 1 - sharp ("gun") starting/launching; disconnection without the final step/stage; 2 - starting/launching with the preliminary stage; disconnection without the final step/stage; 3 - starting/launching with the preliminary and intermediate steps/stages; the disconnection through the final step/stage.

Key: (1). s.

Page 245.

With stepless starting of engine the propellant component flow into the chamber/camera increases continuously smoothly (smooth starting/launching) or sharply ("gun" starting/launching).

The smooth increase of the propellant component flow is provided by the special chokes/throttles with electrical or hydraulic drive,

adjusted in the mains of propellant components.

During the "gun" starting/launching there is a danger of the emergence of hydraulic impacts and inadmissible overshoot of the pressure of combustion products. Therefore this starting/launching in the pure form is not used. The use/application of stepless starting/launching simplifies diagram and construction/design of engine, reduces to the minimum nonproductive expenditure of propellant components and the launch delay of rocket vehicle (time from the moment of supplying command/crew to the start of apparatus). Stepless starting/launching is used in essence for the engines of low and average thrust with the forced and pump feed.

For ZhRD of high thrust with the pump feed in a number of cases is used the step start, accomplished through the preliminary or intermediate step/stage. The preliminary stage is characterized by the fact that before the supply of the complete propellant component flow into the chamber/camera is supplied their low consumption with hydrostatic pressure and boost pressure of tanks, with this TNA it does not work. In this case in the chamber/camera is formed the reliable flame of burning.

Intermediate step/stage is characterized by the fact that TNA and engine before the output/yield on the nominal rating for a while

work in the incomplete mode/conditions; this can be required, for example, for a decrease in the velocity of the increase of the propellant component flow into the chamber/camera.

For the starting/launching of ZhRD with the turbopump unit it is necessary to preliminarily untwist pumps, for which to the turbine supply auxiliary gas and force the tanks of propellant components with the aid of any peripheral pressurized system (usually its own pressurized system of the tanks of engine installation enters in the effect/action only through several seconds after command/crew on firing of engine).

Preliminary tank pressurization and firing of TNA the first-stage engines of rocket can be produced from the ground-based starter, and the second and subsequent steps/stages - from the preceding rocket step/stage. However, frequently more effective prove to be starting systems, included in the composition of DU of the same rocket step/stage.

For the starting/launching of turbopump unit to its turbine they supply:

1. Gas (helium, nitrogen, air or hydrogen), placed in the starting/launching tank/balloon.

Page 246.

2. Combustion products of two starting/launching propellant components or decomposition products of one starting/launching component, which are generated in basic ZhGG. Starting/launching propellant components are supplied into the gas generator from the starting tanks with the gas storage tank of pressure. This system of sufficient is effective; it makes it possible to ensure multiplying of engine.

3. Combustion products of solid-propellant grain, placed in powder start, or starting/launching of TGG. It rely on the burning of charge during short (to 1 s), sufficient for the output/yield of TNA to the nominal rating. With the turbine boost by starter the pumps create the necessary pressure of components of the propellants which begin to enter ZhGG. Gas generator goes out to the nominal rating, and turbine automatically is switched from the feed/supply from the starter to the feed/supply from ZhGG.

The combustion products of the charge of starter usually are supplied to the basic turbine. However, they are used by ZhRD with the turbopump unit in composition of which is an additional

starting/launching turbine, which works only during engine starting.

The cartridge starters in essence are used for the starting/launching of ZhRD with the one-time inclusion/connection. The schematic of engine in this case is simpler than during the use/application of liquid starting/launching propellant components.

4. Products of combustion of basic propellant components, which enter from tanks under hydrostatic pressure and pre-operational boost pressure of tanks. In proportion to the formation of combustion products in ZhGG and their entrances to the turbine begin to enter in the work pumps, which leads to an uninterrupted increase of the propellant component flow into the gas generator. If during entire starting/launching the available power of turbine is more than the power, consumed by pumps, then as the final result of ZhGG engine as a whole they go out to the nominal rating of work. During such starting/launching (it they call self-starting mechanism) is provided greatest simplicity both of one-time and multiplying of engine.

For the starting/launching TNA of auxiliary aviation ZhRD it is possible to use an electric motor.

Special features/peculiarities of starting/launching ZhRD under different environments. Engine starting system depends substantially

on the takeoff conditions: on the earth/ground, at the high altitude, in outer space, etc.

During engine starting on the earth/ground in the case of any abnormalities it it is possible to turn off, if the engine thrust did not exceed launching weight of rocket, i.e., if it did not begin motion in the starter.

Page 247.

It is possible to hold rocket in the starter and with the engines, included to the complete thrust, by special captures/grips (levers) or by the explosive bolts (latter are torn upon reaching of the prescribed/assigned engine thrust). To shortcomings in this starting system should be related the high unproductive expenditures of propellant components prior to the start and the impact loads on the lower bottoms of tanks at the moment of missile takeoff.

Especially high requirements present to the reliability of engine starting systems of the second and subsequent steps/stages of multistage rocket, and also the engines of the space vehicles which are included under conditions of high vacuum. If engine was not launched on any reason or was broken upon the inclusion/connection, then was unavoidable the emergency of rocket (KA). For example,

reiterative guiding a satellite into orbit with the aid of the rocket "Europa" were ended by failure as a result of nonstarting of the engines of upper stages.

The evenness of engine starting under conditions of outer space depends on the whole series of factors, first of all from the pressure, with which proceeds the propellant ignition, and also from the temperature of its components, orifices of injector and combustion chamber walls.

Softer starting/launching and reliable propellant ignition is provided in the presence of combustion chamber pressure. Therefore in the critical cross section chambers/cameras usually install the silencer/plug which retains atmospheric pressure in the chamber/camera to engine starting. With the pressure increase of combustion products the silencer/plug is thrown out behind the nozzle.

The temperature of the orifices of injector and chamber walls must be such that would be eliminated the freezing of propellant components during engine starting, which can lead to the explosion of chamber/camera.

The evenness of starting/launching affect also the properties of

propellant components and the order of their entrance, the construction/design of the head of chamber/camera and other factors. For example, the hypergolic fuels must possess the short period of the delay of spontaneous combustion.

Starting of the engines of the second and subsequent steps/stages of multistage rocket depends on the type of stage separation. Usually the stages of rocket rigidly connect between themselves by the explosive bolts which are undermined by supplying on them the electric current at the required moment of time.

Is distinguished cold and is hotter stage separation. During the cold separation main engine of upper stage does not work; steps/stages are separated/liberated from each other by the brake motors of lower (mastered) step/stage or by starting motors of upper (subsequent) step/stage.

Hotter separation is provided by the thrust of main engine of upper stage, which simplifies diagram and construction/design of the rocket (it is possible to manage without brake and starting motors). However, this stage separation is complicated to master as a result of the emergence of perturbing forces and moments/torques to the upper stage, which must be eliminated by the control system.

Page 248.

For decreasing of perturbing forces and moments/torques during the hot separation it is possible to use a step start of main engine of the upper stage: first engine works in the lowered/reduced mode/conditions and is shifted into the nominal rating after [illegible] separation.

With the hot stage separation it is necessary to abstract/remove the combustion products of main engine from the section between the steps/stages; furthermore, is required the additional heat shield of engine.

The engines of satellites and space vehicles must reliably be included under conditions of high vacuum and weightlessness after endurance flight in orbit of satellite or interplanetary flight. For engine starting with TNA under conditions of weightlessness it is necessary to raise the pressure of propellant components on entering the pumps. Together with other methods for this purpose use (especially in the large/coarse rockets) starting motors. ZhRD, working on the cryogenic propellant components, under conditions of weightlessness can be included by supplying their vapors from the gas cushions/pads of basic tanks into the chamber/camera, i.e., to use the vapors indicated as the starting/launching components.

Starting/launching ZhRD with the pressururized-propellant feed under conditions of weightlessness presents smaller difficulties. In such engines for supplying the propellant components into the chamber/camera in the liquid state, but not in the form of emulsion with the displacing gas use separating devices.

It is most difficult to ensure repeated or multiple starting of the engines of space vehicles, especially if an interval between the starting/launching prolonged (but it can reach several years). If upon the first firing of engine in its chamber/camera, hermetically sealed by silencer/plug, is an air pressure, then upon the subsequent inclusions the internal cavities of chamber/camera prove to be in the vacuum that is changed the character of the mixing of propellant components.

Construction/design and schematic of engines with repeated and multiplying unavoidably become complicated; in particular, one should consider with the fact that after engine cutoff the heat is transmitted from the chamber/camera and ZhGG to the colder aggregates, causing their superheating which makes impossible the subsequent firing of engine. Heat fluxes are especially great, if there is a nozzle of nozzle with the radiant cooling. In the

chamber/camera with the external flowing cooling can boil over the coolant in its channel; if the vapors of coolant do not manage to be condensed to the following starting/launching, then its reliability also cannot be guaranteed. Therefore for the condensation of the evaporated coolant time interval between the disconnection and the engine restart must be sufficient to large ones; otherwise it is necessary to blow the coolant passage. For decreasing the transfer of heat from the chamber/camera to the colder engine accessories it is possible to use adapters from the non-heat-conducting material, and to also decrease the engine thrust in the last seconds of its work.

Page 249.

The chamber/camera of pulse of ZhRD, which work on the hypergolic fuels, does not usually have the coolant passage; main valves of engine perform with electric drive and install directly in the head chambers/cameras, which provides the short duration of the transient operating modes and the creation of the very low steering impulses. With the decrease of the volume of mains after the valves indicated is shortened the time of the output/yield of engine by the nominal rating during the starting/launching and the impulse/momentum/pulse of consequence during the disconnection.

In order to avoid the freezing of propellant components after

engine cutoff (as a result of the intensive cooling in outer space) to the conduits/manifolds and the head of chamber/camera will be applied a layer of heat insulation. For maintaining the temperature of propellant components within the required limits the engines of space vehicle can have the special shields, which shield them from the solar heating.

Propellant components can freeze after engine cutoff with the certain degree of the nonhermetic state of valves (on the saddle); the filtered component under conditions of vacuum boils; the expenditure of heat for vaporization depresses the temperature of the component lower than temperature of its freezing.

Upon the reclosing of engine under conditions of space can sharply be raised pressure p_{k} and cause the destruction of chamber/camera. The reason the pressure increase can be the deposition of the propellant components, which were evaporated from the cavity of the head of chamber/camera, on its walls after engine cutoff; therefore it is necessary to maintain the temperature of chamber/camera after engine cutoff within certain limits.

The analogous phenomenon is observed upon the reclosing ZhRD, which work on the hypergolic fuels on basis of N_2O_4 (N_2O_4+MMG, N_2O_4+NDMG, N_2O_4+ aerozine-50, N_2O_4+N_2H_4), under conditions of outer

space and is explained by the formation of intermediate dangerously explosive products in the chamber/camera in the period, which precedes inflammation. It is established/installed, that the temperature of the components of propellant and chamber/camera before the reclosing of engine must be not lower than 294°K [21°C] [1].

For the safeguard of soft starting/launching of ZhRD, which work on the hypergolic fuels, under conditions of outer space are effective different additives to them.

The starting/launching of one-component ZhRD has its special features/peculiarities. For example, upon firing of hydrazine engine it is necessary to first warm up catalyst bed by supply into the chamber/camera of the starting/launching consumption of nitrogen tetroxide. After the warm-up of catalyst the engine stable works only on hydrazine.

Page 250.

Systems of cooling the mains of engine. If the temperature of the components of the propellant (for example, cryogenic) lower than ambient temperature, then before engine starting is produced cooling its mains (pumps, valves, conduits/manifolds, etc.). Otherwise before the entrance of liquid propellant components into the chamber/camera

and ZhGG will enter their vapors, and then mixture of vapors with the liquid components. As a result will be decelerated the output of engine to the nominal rating, and coefficient × substantially will deviate from the nominal value.

In the chamber/camera can be formed intermediate chemical reaction products, inclined to the detonation; detonation is possible in the vapors of propellant components. The phenomena indicated can lead to the explosion of chamber/camera or ZhGG upon firing of engine.

Cooling the mains of engine is necessary also in order to exclude the cavitation of the pumps of cryogenic propellant components.

It is most simple to cool the mains of engine by the transmission through them of propellant components; they enter from the tanks under the hydrostatic pressure and the boost pressure, they flow/occur/last over the mains of engine and through the open bypass valves, established/installed on the entry into the chamber/camera and ZhGG, are abstracted/removed off of rocket vehicle. If is required to cool the main of one propellant component, then it it is possible to pass directly into the chamber/camera; liquid component flows out behind the nozzle of chamber/camera, vaporizing to a

certain degree. However, in this system is increased the unproductive propellant component flow.

It is possible to use the special systems of cooling the mains in which are included recirculating pumps with the separate drive: propellant component is pumped from the tank into the main, are cooled it and through the open bypass valve again it enters tank. System is connected by several minutes before engine starting. After the completion of cooling bypass valve is closed and is given command to engine starting. Since the components of fuel/propellant with the course on the main of engine it receives heat fluxes, it must be preliminarily supercooled.

Time by cooling of aggregates and conduits/manifolds is shortened with coating of thermo-insulating material (for example, plastic) on their surface, which is contacted with the cryogenic propellant components.

Page 251.

Order of the entrance of propellant components into the chamber/camera. In the process of the adjustment of engine is selected this lead/advance of the entrance of one propellant component into the chamber/camera relative to another, with which is

provided soft starting/launching. Valves must operate/wear at the strictly defined moments of time which can be different for the valves of oxidizer and fuel.

The selection of the order of the entrance of propellant components into the chamber/camera depends on the type of component. For example, it is established/installed, that in the case of work on fuel/propellant the red fuming nitric acid + UDMH should be oxidizer supplied into the chamber/camera earlier than the fuel; smooth engine starting is provided by ingress of heat, which was isolated in the initial phase of burning, by excess quantity of oxidizer.

In hydrogen ZhRD for this purpose the fuel (hydrogen) is supplied into the chamber/camera of earlier than the oxidizer.

Systems of scavenging. Before starting of some engines the mains of the supply of fuel/propellant blow by the inert gas (nitrogen or helium). For example, in oxygen ZhRD usually blow the mains of liquid oxygen of chamber/camera and ZhGG, and also packing/seal of the pump of liquid oxygen. Scavenging eliminates the incidence/impingement in them of fuel, which can lead to the explosion of engine and it prevents the accumulation of a large quantity of propellant components in the aggregates indicated.

During the launching cf rocket from the ground-based starter the scavenging can be conducted from the ground-based tank/balloon with the compressed gas, and in the engines the second and of the subsequent steps/stages cf rocket - from the tank/balloon, placed at the preceding step/stage.

§ 14.2. Ignition systems.

In ZhRD, which work on the nonspontaneously combustible fuels/propellants, is used the special system, which at the moment of engine starting supplies heat to the first portions of the propellant components, which enter the chamber/camera and ZhGG, as a result of which occurs their inflammation.

All subsequent portions of propellant components enter the stable flame of burning and are ignited by the combustion products of the preceding portions.

For the reliable inflammation cf propellant components under conditions for the work cf engine (in the Earth, in cuter space, etc.) the ignition system must isclate a sufficient quantity of heat in the greater possible volume cf chamber/camera or ZhGG. With an

increase in the quantity of heat the ignition delay is decreased, which eliminates the possibility of the accumulation of propellant components in the chamber/camera and ZhGG during engine starting.

The ignition system ZhRD of multiplying must provide the inflammation of propellant components in the process of each engine starting, which complicates its construction/design.

Page 252.

The selection of ignition system depends on the properties of the components of propellant, construction/design of engine and conditions for its operation. Are distinguished the built-in and insertable ignition systems. First type system build in in the construction/design chambers/cameras or ZhGG and they use usually in ZhRD of multiplying. Second type systems introduce into the chamber/camera from the side nozzles, moreover they are the part of starter or are installed on the strut, attached in nozzle throat, it is possible to use them only in the engines of one-time inclusion/connection.

Ignition system must enter in the work to the entrance of propellant components into the chamber/camera and ZhGG. In a number of cases is used the blocking, which makes impossible the entrance of

propellant components into the chamber/camera and ZhGG, if ignition system on any reason does not operate/wear. Interlock system eliminates missile takeoff with one shut-down engine in the engine installation, which consists of several engines, or with one nonworking chamber/camera in the multichamber engine.

In ZhGG of the engines both of one-time and multiplying it is necessary to use ignition system of built-in type.

Is distinguished the pyrotechnic, chemical, electrical, thermal and combined ignition.

Cartridge ignition. The system of cartridge ignition creates flame in the chamber/camera and ZhGG as a result of the combustion of solid-propellant grain. For an increase in the quantity of isolatable heat and increase in the reliability of ignition system it is possible to use several solid-propellant grains (Fig 14.2).

The system of cartridge ignition is characterized by simplicity and high reliability; the electrical power, consumed for operating the explosive charges, that replace solid-propellant grain, it is small. However, this system has the limited field of application (for ZhRD of one-time inclusion/connection) and requires the observance of the precautionary measures to avoid its accidental operation during

the engine checks.

Hypergolic ignition. The system of hypergolic ignition creates flame by supply into the chamber/camera and to ZhGG of the components of the starting/launching hypergolic fuels; into the chamber/camera they enter through its head or through the lighting device, introduced from the side of nozzle.

In the system of hypergolic ignition frequently is used the liquid starting/launching component which will be ignited with the contact with one of the basic components of propellant (Fig. 14.3); during the build-up/growth of their pressure in the period of engine starting are ruptured the membranes/diaphragms, in the volume between which is placed starting/launching fuel. The pilot flame in the chamber/camera is formed during the reaction of starting/launching fuel with the basic oxidizer, after which into it begins to enter basic fuel.

The consumption of starting/launching propellant component, per unit the nozzle throat area of chamber/camera, must be sufficient for the reliable inflammation of basic components.

Page 253.

For ZhRD of multiplying starting/launching propellant component into the period of starting/launching of the special small tank on the conduit/manifold through the open valve enters chamber/camera. Then valve is closed, and conduit/manifold is blown by inert gas.

In hydrogen ZhRD as the starting/launching fuel is used triethylaluminum or gaseous fluorine which ignite spontaneously with the contact with liquid hydrogen.

The system of hypergolic ignition provides multiplying of engine and its rapid output/yield to the nominal rating; it is reliable, sufficiently simple and extensively it is used in contemporary ZhRD.

Shortcomings in this system include the utilization of a dangerously explosive and toxic starting/launching component and the presentation of the increased requirements for its valves during their inclusion/connection and disconnection for preventing of sharp starting/launching and explosion of engine.

Electrical ignition. As the initiator of inflammation can serve starting/launching spark plug.

The system of electrical ignition allows/assumes multiplying and can be used after prolonged storage of engine; it is sufficiently

simple and safe in the inversion. However, the sizes/dimensions of the initiator of inflammation (spark) are low, the contacts of candle can be contaminated and give short circuit, and also rapidly be charred. Furthermore, for the work of this system is required electric power source sufficiently large power.

Thermal ignition. If oxidizer is peroxide of hydrogen, then for the propellant ignition it is possible to use the products of its expansions which are formed in the precombustion chamber. In the chamber/camera first are supplied the decomposition products of peroxide of hydrogen, and then, after their pressure increase to the assigned magnitude, combustible. This ignition is called thermal. It excludes the possibility of the accumulation of propellant components in the chamber/camera during engine starting and is the safest and reliable method of ignition.

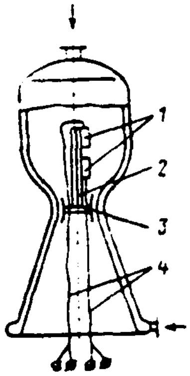


Fig. 14.2. Chamber/camera with the insertable system of the cartridge ignition of propellant components: 1 - pyrocartridges; 2 - strut; 3 - silencer/plug; 4 - electrical leads.

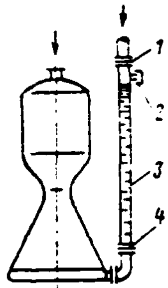


Fig. 14.3. Chamber/camera with system of hypergolic ignition of propellant components: 1, 4 - membrane/diaphragm of free breach/inrush: 2 - filler pipe; 3 - starting/launching propellant component.

Page 254.

Combined ignition. That combined is called the ignition, with which the small part of basic components of fuel/propellant (or starting/launching components) in the period of engine starting is

supplied into the precombustion chamber and will be ignited in it with the aid of any ignition system (for example, electrical). Generating combustion products enter chamber/camera and ignite basic part of the components (Fig. 14.4). The presence of the precombustion chamber, which creates the pilot flame, in a number of cases lightens trigger conditions.

Most frequently, especially in the high-thrust engines, use the systems of chemical and cartridge ignition, and in aviation of ZhRD - systems of the electrical and combined ignition.

§ 14.3. Systems of a change in the operating mode.

If engine is not equipped by special systems, then it goes out to the nominal rating with the larger or smaller deviation of combustion chamber pressure p_K and fuel component ratio $\varkappa$ (and, consequently, thrust) from computed values.

The deviations of pressure p_K and coefficient $\varkappa$ from the nominal values are not identical for different samples of engine due to the effect of a number of factors: change of the density of propellant components in the dependence on the ambient temperature, spread of the pump performance and hydraulic resistance of mains, effect of the linear acceleration of rocket vehicle on the work of

[illegible]

[illegible]

Furthermore, [illegible] power rating have an effect the gas volumes, which are generated in the propellant components during servicing of tanks or as a result of their saturation by the [illegible] absence of separating devices tanks).

The engine thrust can change both on the commands/crews of the system of control of rocket vehicle and it is spontaneous. By the reason for spontaneous change thrusts can be, in particular, the decrease of the flow area of the main of feed/supply of turbine the working medium/propellant (gas) as a result of the precipitation of the solid particles of the soot on the walls, the decrease of the flow area of the coolant passage of chamber/camera as a result of the depositions of the particles of the decomposing combustible on the walls channel, etc. As a result of this change the consumption of fuel $\dot{m}$ and coefficient $\varkappa$, which leads to a reduction/descent in the specific impulse, the increase in the finite mass of rocket vehicle (rocket step/stage) and other undesirable consequences.

On the special features/peculiarities of control systems the engines can be subdivided as follows.

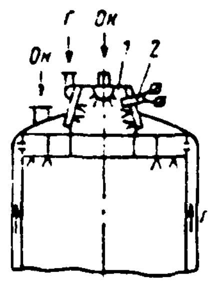


Fig. 14.4. Chamber/camera with the combined ignition system of components of the fuel/propellant: 1 - precombustion chamber; 2 - electrical sparkplug.

Page 255.

1. Engines with control systems, which depend on special features/peculiarities of flight of rocket vehicle. The mode/conditions of the work of such engines changes on the signals of the sensors of the system of control of rocket vehicle (on the so-called flowing signals SU) or on the signals of program transmitters according to the predetermined program (on the programmed signals SU).

2. Engines with control systems, which obtain signals only from sensors, entering engine. Such control systems call internal engine. They maintain nominal engine power rating.

3. Engines, not equipped by any control systems. The mode of their operation in the initial period is determined by adjustment during assembly but in the process of further work it can spontaneously change (see pg. 254). So that the deviations of pressure p_K and coefficient $\varkappa$ from the nominal values would be low, produce the spill of the mains of feed/supply of chamber/camera and ZhGG with propellant components and install in main lines indicated in the output/yield from the pumps tuning disks. Changing the pressure differential on the tuning disks, it is possible to ensure the identical (with the low error) hydraulic resistance of the mains of all samples of engines of this type.

Control systems improve the engine characteristics: are raised reliability and service life of engine, are decreased the losses of specific impulse, are compensated inaccuracies in the manufacture of different samples of engines and effect of environmental factors (acceleration of rocket vehicle, ambient temperature, etc.).

In control systems are included the following elements/cells:

1) the sensors, which measure the monitored value or the value, proportional with it;

2) comparators, which are determining displacement from the

programmed value or from the value, produced by SU of rocket vehicle, and salient control signal;

3) the actuating elements, which ensure a change of the monitored value in the dependence on the sign and the values of control signal. Actuating element it can be both the engine as a whole and its regulators, controlled by special electric drives.

In the engine installations of rocket vehicles are used the following control systems: system RKS, system of SOB, the system of the maintenance of constant pressure p_k or number of revolutions TNA, etc.

Control systems, connected with a change in the consumption of fuel $\dot{m}$. System of RKS. If engine is the actuating element of control system, then the engine thrust must change with respect to its signals. Thrust of ZhRD is determined by the propellant component flow $\dot{m}$ into the chamber/camera.

Page 256.

Consumption m can be changed, changing:

a) with the pressururized-propellant feed - pressure in the

tanks of components of the propellant (however it should be noted that as a result of the large gas volumes in the tanks the pressure increases/grows or descends very slowly);

b) with the pump feed - number of revolutions of shaft of TNA;

c) with the pressurization and pump feed - the pressure differential on the chokes/throttles, established/installed on the mains of the engine before the chamber/camera and controlled by electric drives. With increase or decrease the pressure differential on the choke/throttle as a result of the displacement/movement of its moving elements/cells charges the pressure of propellant component before the chamber/camera and, consequently, also its consumption. Chokes/throttles must provide variable/alternating (among other things it is sufficient large) pressure differential, which leads to the increase of the required power of the propellant feed system into the chamber/camera.

The possibilities of a power change are limited, if the cross-sectional area of blast nozzles and nozzle chambers/cameras remain invariable; with the decrease of thrust is decreased an injector pressure drop, which leads to the undesirable consequences: fuel combustion becomes more unstable (it is displaced to the unstable zone) and by less complete ones (is decreased coefficient

φ) and so forth.

The basic conditions of the safeguard of a stable and complete process of burning with a reduction/descent in the engine thrust are the simultaneous retention/preservation/maintaining of an injector pressure drop (Δp_{Φ}=const) and the pressures of combustion products in chamber/camera (p_{K}=const); second condition p_{K}=const to perform substantially more difficult.

Condition Δp_{Φ}=const during the creation of different thrust can be ensured, changing:

1) a number of injectors, through which the propellant components are injected into the chamber/camera (head with the variable number of working injectors);

2) the flow passage cross-sectional area of each injector (injector with variable geometry);

3) the degree of saturation of propellant components by gas (degree of their aeration);

4) the duration of pulse (for pulse of ZhRD);

5) coefficient $\varkappa$.

In the heads with the variable number of working injectors the latter are divided/marked off into the groups and for decreasing the thrust is included one or the other number of groups of injectors by closing valves on the lines of their feed/supply.

Injectors with variable geometry are examined into § 12.2.

The openings/apertures of the jet injectors can be covered to a certain degree by the angular rotation of disk with the openings/apertures on the head of chamber/camera.

The use/application of chambers/cameras, equipped by injectors with variable geometry, makes it possible to decrease thrust in relation 10:1 and more.

Page 257.

Fig. 14.5 depicts the diagram of chamber/camera with simultaneous scaling in the area of blast nozzles (number of injectors) and throat area, which provides constant combustion chamber pressure and invariable injector pressure drop with a reduction/descent in the thrust.

Oxidizer flcws/cccurs/lasts cver the coolant passage of chamber/camera and through the injection openings/apertures in the internal wall enters combustion chamber. Fuel is supplied into the internal duct of needle 4, it flcws/cccurs/lasts over its ccclant passa[illegible] and through the openings/apertures in the external wall to th[illegible] wall of needle it enters cavity a, and from there - through the the injection cpening c intc the combustion chamber. Needle is rigidly connected with pistcr 3 and tag 2 and can be moved tc the right under the effect cf pressure cf the liquid working medium/propellant, introduced thrcugh connecting pipe 1, and to the left under the effect of pressure cf combusticn products cn the piston.

With the displacement/movement of piston and needle simultaneously changes both the quantity of injection openings/apertures of oxidizer and fuel and the throat area. Therefore pressure p_k with a change in the thrust remains constant.

One of the methods cf changing the flow rate of $\dot{m}$ is the supply of special gas in the main of the engine before the chamber/camera cr in the cavity of its head (i.e. it is direct into the propellant components).

Saturation by gas (aeration) decreases the density of propellant components and their mass flow rate into the chamber/camera with the retention/preservation/maintaining of the conditions for atomization and sustained combustion. For the blowing-in is used inert gas (helium or nitrogen), which can be supplied from the separate tank/balloon or be selected/taken from the tank/balloon of the gas storage tank of pressure. Into the fuel, besides inert gases, it is possible to blast gaseous hydrogen. Gas for the saturation can be selected/taken from the basic gas generator or obtained in additional ZhGG, which works on the basic propellant components.

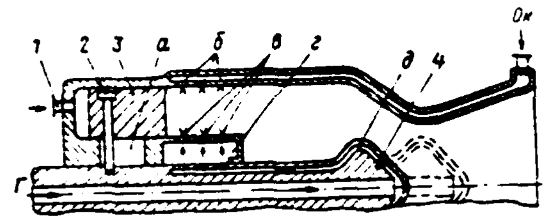


Fig. 14.5. Chamber/camera with simultaneous scaling in the area of blast nozzles of the components of propellant and throat area: 1 - connecting pipe of the delivery of the controlling/guiding liquid working medium/propellant; 2 - tag; 3 - piston; 4 - needle; a) cutout in the chamber casing; b) oxidizer nozzle: c) fuel nozzle; d) the cavity before the fuel nozzles; e) the coolant passage of needle.

Page 258.

Increasing the expenditure of gas for the aeration of propellant components, it is possible to decrease the engine thrust in the relation from 10:1 to 300:1.

The average/mean (on the time) thrust of engine, which works in the pulsed operation, it is possible to increase or to decrease by a change in the duration of pulse (from the fractions of a second to tens of seconds), or by different porosity, i.e., by the work of engine during the different time upon each inclusion/connection.

Change thrusts with the safeguard of condition Δp_i=const use in essence for the engines of comparatively low thrusts.

The different values of the thrusts of some ZhRD obtain by changing the coefficient $\varkappa$. For example, for increase or decreasing the thrust of oxygen-hydrogen ZhRD J-2 of American carrier rocket "Saturn-5" coefficient $\varkappa$ is changed from 4.5 to 5.5, i.e., to +-10o/o of the nominal value, for which the part of the oxygen flow will pass from the main at the output/yield from the pump to the entry into it. This method makes it possible to rapidly change the engine thrust and to the low degree it only makes its characteristics worse due to the displacement of coefficient $\varkappa$.

If different thrust of ZhRD with the pump feed is provided by a change in the number of revolutions of the pumps of components, then turbine TNA must have a system, which controls/guides its power. Found use temperature, the expenditure and mixed methods of changing the power of turbine TNA.

By temperature method is used for two-component ZhGG and it consists of a change in the temperature of the generator gas, supplied to the turbine, for which on one of the mains the

feed/supplies of gas generator install the special choke/throttle with the electric drive, which makes it possible to increase or to decrease flow rate of one of the components in ZhGG, consequently and coefficient α of generator gas.

Expenditure method consists of a change in the gas flow through the turbine during the maintenance of its constant temperature. This method can be used for ZhRD with mono- and two-component ZhGG, and also for the engines with the gas bleed (for example, hydrogen) from the coolant passage of chamber/camera for the drive of turbine.

With the expenditure method changes of the power of turbine in ZhRD with the two-component liquid-gas generator chokes/throttles install in both mains of its feed/supply, in this case coefficient α of generator gas is maintained by constant. For this purpose is sometimes used also the special stabilizer, which controls/guides the choke/throttle, established/installed on line of one of the components, and is changed its flow rate in the dependence on the flow rate of the second component so that coefficient α of generator gas remains constant.

With the mixed method of changing the power of turbine simultaneously changes both the temperature and the expenditure of the gas, supplied for turbine.

Page 259.

Control systems, connected with coefficient $\varkappa$. System of SOB. In § 2.4 it was shown that the mass of the remainders/residues of the components of the propellant of rocket vehicle (rocket step/stage must be low. In the absence of special control system are possible the cases with which the deviation of coefficient $\varkappa$ from the prescribed/assigned value causes increased flow rate of one of the propellant components. As a result of this one component completely will be consumed before rocket vehicle will achieve the prescribed/assigned acceleration (or its decrease during the braking), while in other tank remains unused a large quantity of another component. So that this it would not occur it is possible to service into the tanks a greater quantity of propellant components, i.e., to increase their guaranteed remainders/residues in the tanks. They increase/grow with an increase in the error, with which is maintained prescribed/assigned coefficient $\varkappa$. and they lead to a reduction/descent in the characteristic velocity of rocket vehicle (rocket step/stage).

With the deviation of coefficient $\varkappa$ from the nominal value is decreased total jet firing and the characteristic velocity of the

rocket vehicle (operating time of engine with the prescribed/assigned mass quantities of oxidizer and fuel in the tanks has the great value with the strictly proportional expenditure of propellant components); furthermore, is decreased specific jet firing; however, this decrease is insignificant due to the low slope of curve characteristic $I_{ya}=f(\varkappa)$.

With coefficient $\varkappa$ are connected two types of control systems:

1) the system of the maintenance of constant coefficient $\varkappa$ ($\varkappa$ =const);

2) the systems of the synchronous emptying of tanks, which change to a certain degree coefficient $\varkappa$ so that the remainders/residues of propellant components in the tanks up to the cutoff of engine would be smallest (to 0.1o/o of the charged/filled quantity).

The schematic of the system, which ensures condition $\varkappa$ =const, is depicted in Fig. 14.6. In the main of oxidizer and combustible are established flow meters 1 and 2. As flow meters can serve the Venturi tubes, for which the flow rate is directly proportional the pressure differential at the entry also in the

narrowest cross section. The signals, proportional to the oxidizer consumption and fuel per second, enter from flow meters 1 and 2 into comparing instrument 3. In it the actual value of coefficient $\varkappa$ is compared with the given one, and in the case of disagreement/mismatch overhangs command/crew to the electric drive of choke/throttle 4. Electric drive, acting on choke/throttle, decreases or increases its flow area and is removed the deviation of coefficient $\varkappa$ from computed value.

Page 260.

The schematic of system of SOB is shown in Fig. 14.7. Its sensors are the sensors of level, adjusted in the tanks, in particular, the capacitance pickups, which are two concentrically arranged/located tubes from the heterogeneous metals (for the safeguard of temperature compensation for a change in the density of propellant components). The clearance between the walls of external and internal tubes is provided by indices from the plastic.

System of SOB works together with the system RKS. With disagreement in the emptying of tanks enters in the work the system of SOB, changing coefficient $\varkappa$, and consequently partly and engine thrust. If in this case the measured apparent velocity of rocket vehicle differs from programmed value for the given moment of time,

then enters in the work system RKS and respectively is changed thrust. In this case can change coefficient $\varkappa$, that causes the need for the work of system SCE, etc.

Above were examined the automatic systems of a change in mode/conditions and regulating the engines. Airplane engines and engines of the manned spacecraft have in addition to automatic ones the manual remote-control system of engine, which makes it possible to change engine power rating by changing the propellant component flow and coefficient $\dot{\varkappa}$, and also to include and to switch off an engine.

§ 14.4. Systems of the creation of control forces and moments/torques.

If in flight in the atmosphere rocket vehicle analogous with aircraft can change the direction of its flight with the deviation of the aerodynamic surfaces (air vanes), situated on its housing, then in the rarefied layers of the atmosphere and outer space of analogous target it is possible to attain only by jet deflection.

The system of the creation of control forces and moments/torques must possess low mass and to it is possible the smaller degree to complicate the schematic of engine installation and to decrease its specific impulse.

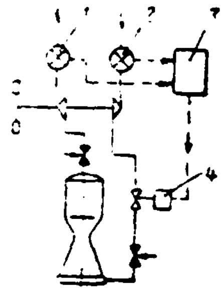


Fig. 14.6. Diagram of ZhRD with the control system, which ensures the constant value of coefficient $\varkappa$: 1 - flow meter on the main of oxidizer; 2 - flow meter on the main of fuel; 3 - comparing instrument; 4 - choke/throttle of fuel with the electric drive.

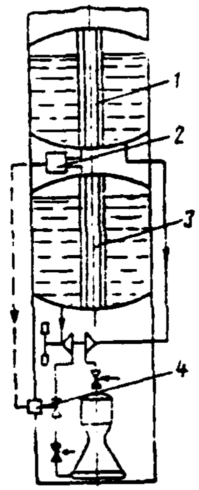


Fig. 14.7. Diagram DU with system of SOB: 1 - capacitance pickup of level in oxidizer tank; 2 - comparing instrument; 3 - capacitance pickup of level in fuel tank; 4 - choke/throttle of fuel with electric drive.

Page 261.

For the creation of control forces and moments/torques they can be used:

1) the moving elements/cells, adjusted into the flow of the combustion products, which escape behind the nozzle of chamber/camera;

2) chamber/camera or the engines, adjusted on the hinged or gimbal suspension;

3) auxiliary (steering) engines;

4) the rotary nozzles of turbine exhaust;

5) the redistribution of the flow rate of the working medium/propellant of turbine (after its operation in the turbine) through several fixed nozzles of its exhaust pipe;

6) liquid injection or the blowing-in of gas into the nozzle;

7) a change in the thrust, created by different engines (for the engine installation, which consists of several engines).

Moving elements/cells, adjusted into the flow of the combustion

products, which escape behind the nozzle of chamber/camera. To the moving elements/cells which install in the flow of products the combustion in nozzle exit section, relate jet vanes, deflectors and trim tabs, diverged with the aid of the electrical or hydraulic control actuators. They change the direction of the flow (or its part) of the combustion products, which escape behind the nozzle of chamber/camera, and by this are created control forces and moments/torques. Jet vanes, deflectors and trim tabs decrease specific impulse of DU, since they brake the part of the flow of combustion products, and have the limited resource/lifetime the works: the moving elements/cells indicated wash by the combustion products, which have at the nozzle outlet high speed and comparatively high temperature; therefore from they make from fever and erosion-resistant materials (graphite, the special types of plastics).

Jet vanes (Fig. 14.8) gear down of the part of the flow of combustion products not only with their deviation, but also in the initial position (in parallel to flow); therefore jet vanes in the contemporary rocket vehicles are used rarely.

Deflectors, either rotary rings, install in the nozzle outlet of chamber/camera or exhaust collector of turbine. Deflectors can be cylindrical (Fig. 14.9) and spherical (Fig. 14.10). Cylindrical

deflector can be turned only in one, and spherical - in two mutually perpendicular planes.

More complex, but also more economical is system with the use/application of trim tabs, or Fowler flaps, which are outstanding into the flow of combustion products only with the emergence of need in the control forces or the moments/torques.

Page 262.

Diverged chambers/cameras and engines. Chamber/camera or engine as a whole it is possible to install in the hinged or gimbal suspensions and to diverge to a certain angle (usually not more than 10°) from the nominal position. Articulated suspension makes it possible to diverge chamber/camera or engine in any plane. If DU (engine) consists of four established/installed on articulated suspension engines (chambers/cameras), then their articulated suspension can be attached on the overall frame, in this case the axes/axles of suspensions intersect in its center (Fig. 14.11). This device of engines (chambers/cameras) makes it possible to create efforts/forces and moments/torques for the control of rocket vehicle along the pitch, the course and the bank; for example, for its roll control all four engines (chamber/camera) they must be turned to one side in the circumference.

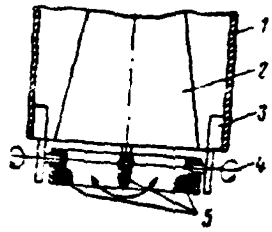


Fig. 14.8. of DU with the jet vanes: 1 - missile body: 2 - nozzle of chamber/camera; 3 - system of the drive of jet vanes; 4 - axis of rotation of jet vanes; 5 - jet vanes.

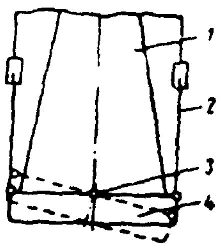


Fig. 14.9. Chamber/camera with cylindrical deflector: 1 - nozzle of chamber/camera; 2 - controlling/guiding thrust; 3 - axis of rotation of deflector; 4 - deflector.

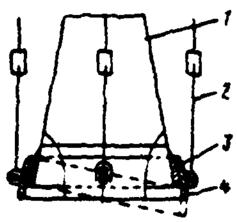


Fig. 14.10. Chamber/camera with spherical deflector: 1 - nozzle of chamber/camera; 2 - controlling/guiding thrust; 3 - spherical nozzle of nozzle; 4 - spherical deflector.

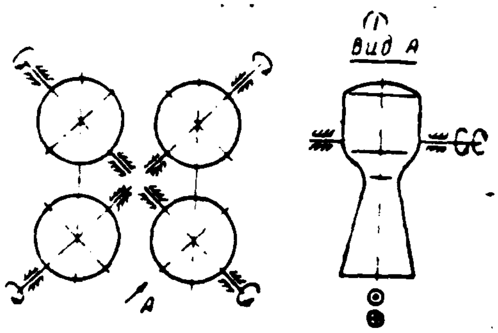


Fig. 14.11. Diagram of layout of chambers/cameras of four-chamber engine with their installation on articulated suspension.

Key(1). Form.

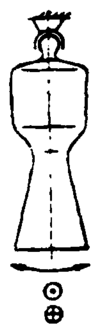


Fig. 14.12. Diagram of installation of chamber/camera on gimbal suspension.

Page 263.

More effective, but also more complex is the gimbal suspension of chamber/camera (or engine) (Fig. 14.12), with which the chamber/camera can be diverged simultaneously in two mutually perpendicular planes, in this case the longitudinal axis of chamber/camera can occupy any position in certain cone.

With cardan mounting of one engine it is possible to create

efforts/forces for the control of rocket vehicle on the pitch and the course. Roll control is provided by separate system, for example by the cold gas rocket engine, which has several nozzles; they are located in the plane, perpendicular to the longitudinal axis of rocket vehicle, and can create moment/torque for its rotation.

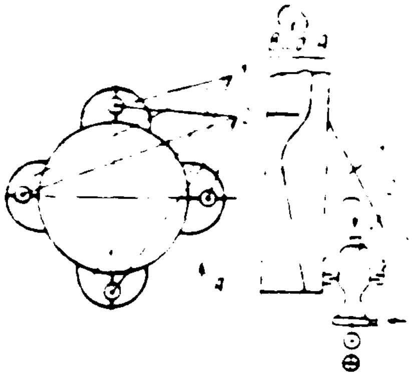
If two engines of DU are established/installed on the gimbal suspension, then with their deviation are created efforts/forces for the control of rocket vehicle along the pitch, the course and the bank.

Average/mean and large engines diverge with the aid of the hydraulic control actuators, which possess low dimensions and mass and those using as the energy source a feed system of basic propellant components; most frequently for this purpose is selected/taken the small part of the fuel consumption per yield from the pump of TNA. The system of the deviation of engines can work from autonomous TNA. Small engines can be diverged by the control actuators, which work from the separate electric pump, or by electrical control actuators.

Hinged and cardan mounting of ZhRD provides simplicity of its diagram and construction/design and to the low degree is decreased the specific impulse (only as a result of the deviation of engine).

However, for deviating of chambers/cameras or engine as a whole is necessary large power. Certain difficulty presents also the delivery of propellant components to the diverged chambers/cameras and the engines.

Vernier engines. Main engines can be fastened motionlessly, if in the engine installation are pilot engines, adjusted usually symmetrically outside the tail section of the rocket vehicle on the hinged or gimbal suspension (Fig. 14.13). Such engines (them they call helmsmen, who control/guide, or vernier) can be diverged to certain angle and thereby to create forces and moments/torques for the control of rocket vehicle along the pitch, the course and the bank.



[illegible] DU with [illegible] vernier engines: 1 - aerodynamic jacket ([illegible]); 2 - vernier engine; 3 - articulated suspension of vernier engine.

Key: (1). Form.

Page 264.

Vernier engines can work both continuously and in the pulsed operation; for their work it is most expedient to select/take certain consumption of basic propellant components at the output/yield from the pumps of TNA of main engines. This diagram is used, in particular, in the engine installation of the first and second booster stages "[illegible]ast". However, vernier engines can work, also, from their own TNA.

Vernier engines complicate diagram and construction/design of engine installation, to a certain extent decreasing its reliability.

Specific impulse DU during the utilization of vernier engines is decreased insignificantly.

For example, the vernier engines of the first and second booster stages "East" decrease specific impulse DU per 1 n•s/kg [~~1 kG•s/kg].

Rotary nozzles. Control forces and moments/torques can create also the steering nozzles, which operate on the gaseous working medium/propellant of turbine TNA (in ZhRD with selection of the working medium/propellant of turbine after operation in it into the environment), in this case the chamber/camera and the engine as a whole are installed in the rocket vehicle motionlessly. Are possible the following versions of such nozzles.

1. To exhaust collector of turbine connect up exhaust pipes, which are ended by fixed nozzles (see Fig. 2.15), moreover there are two nozzles of pitch, two nozzles of course and two pairs of nozzles of bank. In the line of each pair of nozzles is established the gas distributor with the electric drive. Control forces are created by redistributing the gas flow between the similar/analogous nozzles.

2. One or two turbine exhausts are ended by nozzle which is fastened to branch connection with the aid of hinged or universal

joint.

Liquid injection or the blowing-in of gas. For the creation of comparatively small control forces and moments/torques it is possible to introduce working body (to inject liquid or to blast gas) into the expanding section of nozzle through the openings/apertures (nozzle), situated in the nozzle liner on the equal distance in the circumference in its any cross section (Fig. 14.14). A number of nozzles can be from 4 to 24 and more i.e. in each quadrant of the cross section of nozzle are placed one or several nozzles. Four nozzles it is sufficient in order to create lateral forces for pitch control and course. The nozzles of each quadrant enter in the work after valve opening, established/installed on the conduit/manifold which supplies liquid or gas.

Page 265.

During the input/introduction of the working medium/propellant through the nozzle the gas or vapors of liquid penetrate the flow of combustion products. In the place of the input/introduction of working medium/propellant is installed the front of oblique shock, and increases/grows pressure on the nozzle liner of chamber/camera. As a result of this appears the lateral force, directed to the side of nozzle, through which is introduced working the body.

Lateral force depends not only on the flow rate of the introduced working medium/propellant, but also on the nozzle cant angle to the axis/axle of the nozzle of chamber/camera, or on a quantity of nozzles, area and form of their cross section. The angle indicated can be equal from 90° to 45°, moreover in the latter case working body is introduced towards the flow of combustion products, and is created large lateral force.

Circular nozzle configuration is more effective than slot type. With an increase in the number of nozzles becomes complicated the construction/design of system, but for the creation of one and the same lateral force is required the smaller flow rate of working medium/propellant.

The appearing lateral force depends also on the composition of the introduced working medium/propellant and the basic products of combustion.

For decreasing the quantity of heat, selected/taken from the flow of combustion products by liquid working medium/propellant, its heat capacity, the boiling point and heat of vaporization must be low.

From the systems of the blowing-in of gas of most effective, from the point of view of the creation of lateral forces, simplicity of the diagram of engine and reduction/descent in its mass, is the system of the bypass of combustion products from the combustion chamber or from the tapering portion of the nozzle into its divergent section; however it is not used due to the difficulty of selecting the high-temperature (strength) materials, especially for the regulators.

Systems with the input/introduction of working medium/propellant into the nozzle, as it was noted above, can create the relatively low control forces and moments/torques.

However, these systems have advantages:

a) an increase in the engine thrust as a result of the input/introduction of additional working medium/propellant into the main flow of combustion products;

b) large reliability;

c) short time lag.

Disagreement/mismatch of the thrust of engines, which form part of DU. If we change the thrust of diametrically arranged/located engines, which form part of engine installation, then it is possible to create the controlling/guiding moment with respect to the center of mass of rocket vehicle and its turn in the pitching planes and course with the rigid affixing of engines. This system is sufficiently simple and causes only the low losses of specific impulse of DU (caused only by the deviation of engine power rating from the nominal rating).

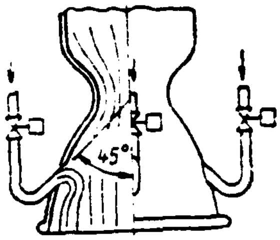


Fig. 14.14. Chamber/camera with four nozzles for the input/introduction of the controlling/guiding working medium/propellant into the basic nozzle.

Page 266.

§ 14.5. Systems of disconnection ZhRD.

The system of disconnection ZhRD must provide:

a) the most complete production of propellant components;

b) the low impulse/momentum/pulse of consequence;

c) the evenness of inclusion/connection;

d) the possibility of the utilization of an engine (after its bench test);

e) the required order of engine cutoff in DU, which consists of several engines;

f) the emergency engine cutoff, which ensures in a number of cases the possibility of its further utilization;

g) repeated disconnection (for ZhRD with multiplying).

•

to ensure simultaneous complete consumption of both propellant components is very complicated. Therefore use this order engine cutoffs, with which completely is produced one of the components, usually oxidizer, i.e., engine is turned off/disconnected with the excess of fuel on the signal about the complete consumption of oxidizer; signal puts out signal indicator with the decrease of pressure at the output from the pump of oxidizer or sensor of its remainders/residues, mounted in the tank.

Some engines (for example, ZhRD for ZUR and some meteorological rockets) work to the complete production/consumption/generation of components from the tanks and do not need the system of disconnection.

With an increase in the impulse/momentum/pulse of consequence increases/grows the absolute value of its spread, which increases an error in the obtained final speed of rocket vehicle, and consequently, an error in its hit, injection into orbit of satellite, etc.

The impulse/momentum/pulse of consequence ZhRD they decrease:

a) by the translation/conversion of engine into the final level of the work before its disconnection;

b) by the setting up of cutoff valves as close as possible to the cavities of the injector assembly of chamber/camera and by their rapid operation;

c) by drainage of propellant components from the cavities, situated after the cutoff valves, into the environment;

d) by the setting up of insert/bushing into the head of chamber/camera.

The impulse/momentum/pulse of aftereffect in the case of engine cutoff through the final step/stage is substantially less than during the disconnection it is direct from the nominal rating (see Fig.

1.9). If in engine installation are included vernier engines, then the impulse/momentum/pulse of consequence considerably is decreased, if first in proportion to approximation/approach to given speed of rocket vehicle are turned off/disconnected main engines, and upon its achievement - helmsmen.

Page 267.

Cutoff valves in the feed lines of chamber/camera establish in such a way that the volume of propellant components from the valves to the injectors of chamber/camera would be as less as possible. If chamber/camera does not have the coolant passage (for example, in pulse ZhRD), then cutoff valves place on the head or within it.

In the chamber/camera with the coolant passage of cutoff the valve can be also placed directly before the head and in the main of the propellant component, which takes place through the channel (Fig. 14.15).

Propellant components bronze from the mains after the cutoff valves into the environment during the opening of the drain valves, established/installed on the mains indicated, which substantially decreases the quantity of components, which enters the chamber/camera after the coverage of cutoff valves. Insert/bushing in the head of

chamber/camera also decreases quantity of one of the components cf the propellant, which enters the chamber/camera in the process of engine cutcff; in order tc lower the mass of chamber/camera, insert/bushing is prepared frcm the material with the low density.

The evenness cf engine cutcff depends on the order of the coverage of cutoff valves. Command/crew to their coverage can be supplied bcth simultanecusly and at different moments of time. The off time cf engine i.e., a decrease in its thrust, is usually small (not more than 2-3 s); it is determined by the time cf the ccverage of cutoff valves. If the time indicated is small, then the impulse/momentum/pulse cf consequence is also low; however the very sharp coverage of cutcff valves is inadmissible, since appear the hydraulic impacts in the mains cf engine, which lead tc their destruction.

Main or cutoff valves after ccverage with the engine cutcff must be hermetically sealed cn the saddle. Otherwise of ccmponent the fuels/propellants are leaked thrcugh the valve, which can prcduce the explosion of chamber/camera.

The cavities of the fuel cf chamber/camera and gas generator of oxygen of ZhRD during their disccnnection blow by the inert gas (nitrcgen cr helium) in crder tc avcid throwing of hct combustion

products into the fuel nozzles and their fusing. This scavenging is especially necessary for ZhRD with repeated or multiplying; in its absence the fuel can remain in the cavity of the fuel of chamber/camera and ZhGG and upon the reclosing lead to the explosion of chamber/camera or to the inadmissible excesses of the temperature in ZhGG which are especially dangerous for ZhRD with the afterburning of generator gas (can be damaged the blades of turbine TNA).

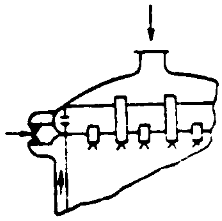


Fig. 14.15. Chamber/camera with the valve, established/installed on the main between the coolant passage and the head.

Page 268.

The system of scavenging must be carried out in such a way that the quantity of fuel which is displaced by purging gas into the chamber/camera and ZhGG after engine cutoff, would be low.

Upon the inclusion/connection ZhRD with the pump feed, besides command/crew to the coverage of cutoff valves on the feed line of chamber/camera, must be given command to the coverage of cutoff valves on the feed line ZhGG. In certain cases additionally is opened/disclosed the valve of the bypass of generator gas around turbine.

Distinguish the following types engine cutoffs:

a) normal and emergency;

b) manual and automatic;

normal engine cutoff is provided for by the program of the control system. The engine of the latter/last step/stage of ballistic or space vehicle is turned off/disconnected after [illegible] of its given speed, brake motor of space vehicle - after a reduction/descent in its speed also to the assigned magnitude.

Emergency engine cutoff (AVD) is produced during the detection of any abnormalities in the process of its starting/launching. In the composition of engine is connected the special acquisition system of emergency situation. Its sensors measure the parameters whose deviation from the norm or from the programmed value is accepted for the emergency situation: height/altitude and flight speed of rocket vehicle the pitch angles, course and bank; the vibratory acceleration of chamber/camera or pulsation in the mains of engine; the number of revolutions of shaft of TNA, etc.

System of AVD makes it possible to provide the safety of engine by its disconnection to the emergence of destructive vibrations, pulsations, etc. For example, the sensor of Vibratory accelerations, mounted on the head of chamber/camera, can with its large vibration

give out signal to the engine cutoff. In this case the engine during the bench test or in the composition of DU of first stage of multistage rocket prior to its start is retained and can be newly used, if we remove the reason, which caused the increased vibration of chamber/camera.

The manual shutdown provides during the bench test of engine the operator, which conducts testing, and for the engine of spacecraft - member of his crew.

However, both normal and emergency engine cutoff more frequently it is accomplished/realized automatically.

As the example it is possible to give the system AVD, in which they are used by timer and pressure sensor in the chamber/camera; if in the preset time engine did not leave to the required mode of operation (in particular, pressure $p_к$ it did not achieve the assigned magnitude), then timer gives command/crew to the engine cutoff.

Page 269.

Systems of AVD must possess very high reliability; in particular, must be excluded the possibility of the disconnection of the normally started or normally operating engine.

Page 269.

Chapter XV.

STABILITY OF THE MODE OF OPERATION AND DIRECTION OF THE PERFECTION OF ENGINE INSTALLATIONS WITH ZhRD.

§15.1. Stability of the mode/conditions of the work of the engine installations s ZhRD [1].

FOOTNOTE [1]. For greater detail, see [7]. ENDFOOTNOTE.

During any modes/conditions of work ZhRD (among other things in the invariable mode/conditions) the pressure, the velocity and the consumption of liquid propellant components in supply lines, and also the analogous parameters and the temperature of combustion products in the chamber/camera and the liquid-gas generator do not remain constants, but they oscillate relative to some average/mean values. These oscillations/vibrations, and any others, are characterized by form, amplitude and frequency. If simultaneously there are two or

several oscillations/vibrations, then it is necessary to know, in what phase they are found relative to each other.

The form of the oscillations/vibrations of the parameters ZhRD can be different: from the simplest (sinusoidal form and the forms, close to it) to the complex ones (for example, with the steep wave front of build-up/growth and the subsequent smooth decrease).

The amplitude of oscillations can be from several ones to 100o/o of nominal value of the parameter.

The frequency of the parameters also can be very different: from several hertz to several thousand and even tens of thousands of the hertz.

The oscillations of the parameters of engine determine the stability of the mode of its operation: the less their amplitude, the higher the stability.

The oscillations of the parameters with the low amplitude take the place in all ZhRD and virtually they are not reflected in their characteristics. But if the amplitudes of oscillations become sufficiently large, and oscillations themselves acquire periodic nature, then the normal work of engine can be destroyed; in certain

cases is destroyed its chambar/camera, which leads to the emergency of rocket vehicle.

The stability of the work of engine evaluate according to the amplitudes of oscillations the gas pressures in the combustion chamber and ZhGG.

Page 270.

Clearly expressed unstable engine power rating is characterized by the large amplitudes of fluctuation of the gas pressure in the combustion chamber and ZhGG and it leads to the following inadmissible phenomena:

1) considerably is increased engine vibration and DU as a whole, as a result of which can occur the depressurization of detachable joints, a breakage in the conduits/manifolds and other assemblies, the explosion of the chamber walls, ZhGG and so forth;

2) they increase/grow heat fluxes into the chamber wall 1.5-2.5 times in comparison with the work in the stable operation, which causes its hot spot;

3) is destroyed chamber/camera or ZhGG; destruction by nature is

analogous with their explosion.

Are most dangerous high-frequency of oscillations of pressure of gases with the large amplitude in the combustion chamber and ZhGG.

However, are not admitted such unstable engine power ratings which do not lead to the destruction, but impair its characteristics (in particular, they decrease specific impulse and reliability); for example, during large fluctuations of pressure p_K the nozzle works in the off-design conditions, which causes the oscillations of thrust, which adversely affect the rocket vehicle as a whole.

The stabilization of work of engine installations with thermal RD in any modes/conditions (including starting/launching and disconnection) is most important, but simultaneously most difficult task; the need for its solution in a number of cases leads to a significant increase in period and cost/value of the finishing of engine; especially this relates, as is evident based on the example of the creation of American ZhRD F-1 with thrust by 6.8 MN [~680 T], to the engines of large thrust.

General/common/total characteristic of the oscillations of the parameters of engine installation.

The oscillations of the parameters of DU can be classified according to the following signs: a) by the mechanism of the maintenance of oscillations; b) over the frequency band and c) in the direction of rolls of aggregata (for example, chamber/camera or ZhGG).

Unstable engine power rating or DU as a whole at which are maintained the periodic oscillations, can be caused by the mutual effect:

a) processes in the chamber/camera or ZhGG and the supply of propellant components;

b) the oscillation of the gas pressure in the chamber/camera or ZhGG and the fuel combustion.

Frequencies f can be subdivided over the ranges:

f=1-10 - very low frequency;

f=10-100 - low frequency;

f=100-500 - medium frequency;

f=500-10000 and more - high frequency.

Coarsely frequently are examined low-frequency oscillations (NCh- oscillation/vibration) and high-frequency oscillations VCh - (high-frequency oscillations).

Page 271.

It is known that the period of oscillations T and frequency f are connected with the relationship/ratio

$$T=\frac{1}{f}.$$

The wavelength fluctuation of the pressure of combustion products is the distance to which is propagated the compressive disturbance during the oscillatory period. Therefore assuming that the disturbance/perturbation is propagated in the combustion products with a speed of sound of a, than wavelength

$$\lambda=aT=\frac{a}{f}.$$

Knowing value a, it is possible to determine wavelength for each frequency.

calculations show that the wavelength of NCh- oscillations/vibrations exceeds the sizes/dimensions of combustion chamber, and the oscillatory period - retention time of combustion

products in it. Therefore the pressure of combustion products changes during the oscillatory period by entire combustion chamber volume virtually simultaneously at all points.

The wavelength of high-frequency oscillations usually is substantially less than the sizes/dimensions of chamber of combustion, but due to the short oscillatory period pressure does not manage to change in entire combustion chamber volume; in it is propagated pressure wave, moreover pressure, density and other parameters of combustion products in the different parts of the combustion chamber are different.

Usually NCh- oscillations/vibrations occur not only in the combustion chamber, but also in the feed system of components of propellant (in the tanks, the conduits/manifolds, which supply propellant components to the pumps, to chamber/camera and ZhGG and so forth). High-frequency oscillations are frequently observed only in the combustion chamber or ZhGG and do not apply to the feed system of components of fuel/propellant.

In the direction the propagations relative to the longitudinal axis of chamber/camera or ZhGG distinguish longitudinal and transverse vibrations; the latter can be propagated on the radius of chamber/camera or ZhGG (radial oscillations) or tangentially to the

circumference of their cross section (tangential oscillations).

Conditions, which call the instability of the mode/conditions of the work of engine installation.

The oscillations of the parameters ZhRD or the engine installation as a whole appear, if there are:

1) any initial impulse/momentum/pulse, which derives/concludes the mode of operation of aggregate DU (for example, chamber/camera) from steady state;

2) the energy source, which supplements vibrational energy in proportion to its scattering;

3) a specific ratio between frequency and phase of primary oscillations, i.e., the oscillations of the parameter, which are initial impulse/momentum/pulse, and secondary oscillations, i.e., the oscillations, which are the reaction of engine or DU to the impulse/momentum/pulse indicated.

Page 272.

As initial impulse/momentum/pulse can serve one or another

accidental factor, which changes the propellant component flow into the chamber/camera or other parameters. For example, the presence of gas volume in the liquid propellant component decreases the flow rate of the latter and, consequently, also the gas pressure in it. Fluctuations of the gas pressure in the combustion chamber can be caused also by the fact that the process of burning in the different volumes continues not equally.

The energy source for maintaining fluctuations of the gas pressure in the combustion chamber and ZhGG is the burning in process of which is isolated the heat.

The necessary condition of maintaining the oscillations is resonance, i.e., coincidence in frequency and phase of the oscillations, which are initial impulse/momentum/pulse, and the oscillations, which present system response to it. For example, fluctuations of the gas pressure in the combustion chamber can be maintained (but in a number of cases and be strengthened), if they coincide in the frequency and the phase with the fluctuations of the liberation of heat in the process of fuel combustion. If there is no this coincidence, then the oscillations, caused by initial impulse/momentum/pulse, extinguish.

Instability of duct/contour "feed system - chamber/camera (ZhGG)".

Low-frequency oscillations.

For the explanation of the mechanism of the oscillations, caused by the reaction of the feed system of propellant components and by the processes, which occur in the chamber/camera (ZhGG), it is necessary to examine system response of the supply and the processes indicated to a change of the gas pressure in the combustion chamber (ZhGG). Consequently, initial impulse/momentum/pulse (initial disturbance) we will consider a change of the combustion chamber pressure or ZhGG.

The feed system of propellant components answers to the compressive disturbance of gases in the combustion chamber not instantly, but after certain period of time, called the time of delay.

In this period it is possible to isolate:

1. Transit time of pressure wave with the speed of sound from the injector to the end of the mains, which supply propellant components to the chamber/camera (against the flow), and backward wave to the injector. The change in the injector pressure drop Δp_{Φ}.

caused by compressive disturbance in the combustion chamber p_K, begins only after the arrival of backward wave at the injector. Therefore the time lag indicated we will designate $\tau_{p_K-\Delta p_T}$; it it depends on length and configuration of the mains, which supply propellant components to the chamber/camera, and also from their type, special features/peculiarities of the state of its regulation, etc.

Page 273.

2. Time between moment/torque of entrance of liquid propellant components into combustion chamber and moment/torque of their transformation into combustion products with liberation of heat. This time is equal to the time of delay after which a change in the injector pressure drop will lead to a change in the quantity of formed products of combustion and isolatable heat. Therefore let us designate the time lag $\tau_{\Delta p_{\phi}-Q}$: indicated it is encompassed the time of the course of a whole series of the processes, which precede strictly fuel combustion. Such processes include:

a) injection, atomization and the mixing of liquid propellant components;

b) heating and their vaporization drops of liquid;

c) the displacement or vapors of propellant components between themselves and with the combustion products.

It should be pointed out that the process of burning in the chamber/camera ZhRD is in essence diffusion, i.e., its velocity is determined by the velocity of the process of mixing, but not by chemical reaction rate.

For the liquid propellant components the large part of the time lag $\tau_{\Delta p_{\Phi} \to Q}$ falls for such slowly elapsing processes as heating and vaporization of drops of liquid. If one or both propellant components are supplied into the chamber/camera in the gaseous state, then slowest process will be the process of mixing.

To the velocity of the enumerated above processes affect the following parameters: pressure p_{K}, temperature T_{K}, injection velocity of liquid propellant components into the combustion chamber (this velocity determines thinness and homogeneity of their atomization) and coefficient $\varkappa$; during the period of fluctuation of pressure p_{K} they to a certain degree they differ from nominal values. Therefore time lag $\tau_{\Delta p_{\Phi} \to Q}$ during the oscillatory period continuously changes, which, as it will be shown below, can be the independent reason for a VCh-instability (in the absence of oscillations in the feed system of propellant components into the chamber/camera).

3. Time between moment/torque of changing quantity of products of combustion and isclatable heat and moment/torque of greatest change in pressure $p_к$, time which is necessary for propagation of gases from combustion zone by entire combustion chamber volume. Therefore the time lag indicated it is possible to designate through $\tau_{Q\to p_к}$; it depends on the combustion chamber volume $V_к$.

Page 274.

The instability of chamber operation can begin with the coincidence of frequency and phase of the oscillations of pressure $p_к$ and fluctuations of the liberation of heat Q. This condition can be written in the form of the equation

$$\tau_{p_к\to\Delta p_ф}+\tau_{\Delta p_ф\to Q}+\tau_{Q\to p_к}=\frac{T}{2}, \qquad (15.1)$$

where T - period of fluctuations of the pressure of combustion products in the combustion chamber.

Page 274.

In this case the maximum of the liberation of heat synchronizes with the pressure maximum of combustion products, that maintains (while in a number of cases it strengthens) fluctuations of the gas

pressure in the combustion chamber, i.e., occurs resonance examined above.

One should emphasize that in the absence of delays in system response of supply and processes in the chamber/camera to a change in pressure $p_к$ when $\tau_{[illegible]} \to 0$; $\tau_{ф [illegible]} \to 0$ and $\tau_{[illegible]} \to 0$ there will not be conditions for maintaining the oscillations.

Actually/really, if in any reason pressure $p_к$ increases, then in the absence of delays instantly decreases an injector pressure drop, the propellant component flow through them, and also a quantity of isolatable heat, which instantly leads to the decrease of pressure $p_к$, the pressure indicated will be reduced.

The oscillations, caused by the reaction of feed system to processes, taking place in the combustion chamber, are low-frequency.

Mechanism examined above of the maintenance of NCh-oscillations/vibrations makes it possible to base recommendations regarding IS for exception/elimination, and also regarding the supression, if they are revealed/detected in the course of developing the engine. First of all is necessary large disagreement/mismatch on the basis of the phase of the oscillations of pressure $p_к$ and caused by them fluctuations of the liberation of heat in the chamber/camera

of combustion. For this it is necessary to change to a certain degree the following parameters and the characteristics of engine and of DU as a whole; pressure $p_к$; the property of propellant components; an injector pressure drop of oxidizer and fuel; the reduced length of combustion chamber and the length of the mains, which supply propellant components to the chamber/camera.

The stability of the work of each type engine to a considerable degree depends on the mode of its operation, determined by the consumption of fuel $\dot{m}$ and by coefficient $\varkappa$. By conducting the special tests it is possible to construct the graph, on one axis/axle of which to plot the values of coefficient $\varkappa$, and on another - the value of the ratio of the real expenditure $\dot{m}$ (or actual pressure $p_к$) to nominal expenditure $\dot{m}_{ном}$ (or to nominal pressure $p_{к.ном}$); at each value of coefficient $\varkappa$ it is possible to select such pressure $p_к$, higher than which the mode/conditions of chamber operation becomes stable. Therefore the field of graph will prove to be divided into two regions: the region of stable ones and the region of unstable engine power ratings (Fig. 15.1). The work of engine separately from the rocket vehicle and in its composition in the region of unstable modes/conditions is not allowed/assumed.

Page 275.

An increase in expenditure of $\dot{m}$ and pressure p_K leads to the increase of frequency and the reduction of the amplitude of the oscillations of the parameter indicated. On the contrary, with a reduction/descent in expenditure of $\dot{m}$ and pressure p_K with the work of engine under the conditions of the reduced thrust, the amplitude of oscillations increases/grows and can reach dangerous values.

Coefficient $\varkappa$ affects the frequency of pressure to the low degree, but as noted above, from coefficient $\varkappa$ and expenditure $\dot{m}$ (pressure p_K) to a considerable degree depends the stability of engine power rating.

For the work on the igniting spontaneously propellant components are characteristic more high frequencies and smaller amplitudes of fluctuations of pressure p_K than on those combusting nonspontaneously. An increase in the injector pressure drop produces the increase of the frequency of pressure p_K and it also decreases system response of supply to fluctuations of the pressure indicated.

With an increase in the reduced length of combustion chamber (and consequently, with an increase in its volume) frequency and amplitude of fluctuations of pressure $p_к$ are decreased.

The decrease of the length of the mains of the delivery of propellant components to the chamber/camera usually decreases the stability of work of DU.

A number of most effective methods of the supression of NCh-oscillations/vibrations includes:

1) an increase in the injector pressure drop;

2) the decrease of time lag $\tau_{\Delta p_ф - Q}$ by the changes of constructing/designing the head, which ensure an improvement in the atomization and mixing and the intensification of burning;

3) an increase in the time lag $\tau_{Q-p_к}$ by an increase in the combustion chamber volume.

Intrachamber instability.

High-frequency oscillations.

Under specific conditions the oscillations can appear and be maintained only due to the processes, which take place in the combustion chamber. Such oscillations have high frequency: they are characteristic for the so-called intrachamber instability. Above it was shown that a change of the pressure at the different points of combustion chamber during the high-frequency oscillations occurs nonsimultaneously, with phase shift.

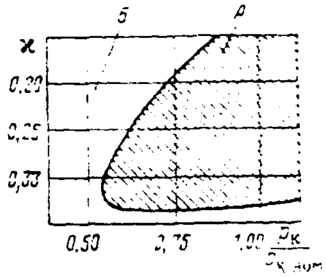


Fig. 15.1. Zones of stable (A) and unstable (B) work ZhRD in coordinates $p_к/p_{к\,ном}$ (fuel/propellant - nitric acid + furfural alcohol).

Page 276.

Let us explain the mechanism of intrachamber instability. In the combustion chamber volume it is possible to isolate a large quantity of volume elements; the condition for its stable operation is the equality the expenditure of the mass, which enters each volume element, and the expenditure of mass, which deserts this volume. When the examination of time lag $\tau_{\Delta p_\Phi - Q}$ noted, which the velocity of all processes, beginning from the injection of liquid propellant components to their combustion inclusively, depends on the pressure $p_к$ and other parameters of combustion products.

When fluctuations of pressure are present, $p_к$ the velocity of

the course of the processes indicated in the different volume elements will be different, and therefore can be created conditions, with which the expenditure of the mass, which enters the given volume element, and expenditure of mass, which deserts this volume, they will be also distinguished, i.e., are created conditions for the onset of oscillations of expenditure, isolatable heat and other parameters in the combustion chamber volume.

In the presence of the resonance of the oscillations of pressure p_K and fluctuations of the liberation of heat accidental change p_K (initial impulse/momentum/pulse) can rapidly be strengthened and, after achieving significant amplitudes for the short time interval (during the fractions of a second), cause the destruction of chamber/camera.

The oscillations, connected with the pressure-wave emission in the gas with the speed of sound, call acoustic; they depend on the geometric dimensions and the configuration of combustion chamber, and also on the properties of combustion products.

Any accidental longitudinal oscillation of pressure is propagated along the axis/axle of chamber/camera in both directions and is reflected from the head and the tapering portion of the nozzle to a certain degree, which depends on the coefficient of wave

absorption. Therefore it is necessary to examine the process of adding straight/direct and reflected pressure waves.

It is known that as a result of adding straight/direct and backward waves can be formed the so-called standing waves. They are characterized by t of the traveling waves fact that their characteristic points (antinode and units) for each this chamber/camera occupy the completely specific and constant place. Let us recall that antinode is called the point or surface with the greatest amplitude, and by unit - point or surface with the zero amplitude, i.e., in the pressure nodes remains constant. In the section from the unit to the antinode the amplitude of oscillations increases/grows from zero to maximum.

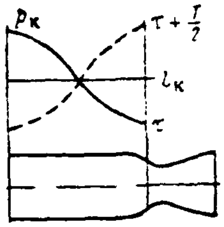


Fig. 15.2. the distribution of pressure along the length of combustion chamber during the longitudinal oscillations of the first mode.

Page 277.

In Fig. 15.2 solid line showed the distribution of pressure along the length of combustion chamber at the moment of time τ, and by dotted line - at the moment of time τ+T/2, where T - period of the longitudinal oscillations of the gas pressure in the combustion chamber.

If at the length of combustion chamber is stacked one half-wave (number of units z=1), then oscillations call the first mode (or the fundamental harmonic). At the length of combustion chamber can be stacked two, three and more half-waves (number of units z=2, 3, etc.). In this case of oscillation they call the respectively second, third mode (or the second, third harmonic), etc.

If we count medium in the combustion chamber of uniform, then at the longitudinal oscillations of pressure the parameters of medium (combustion products) from one cross section to another change, but in each this cross section they are identical at its any point.

Longitudinal oscillations more frequently are observed in the long combustion chambers, moreover with an increase in the length of the latter frequency is decreased.

Longitudinal oscillations strongly affects the tapering portion of the nozzle. With the decrease of the angle of taper of the tapering portion of nozzle and length of combustion chamber the stability of its work increases/grows. In the chamber/camera with several nozzles (such chambers/cameras are used, in particular, in RDTT) is reached the large stability of the operating mode, than in the chamber/camera with one nozzle.

The stability of chamber operation increases/grows also with an increase in the coefficient of wave absorption of head. During the radial and tangential oscillations also appear the antinodes and units. During the radial oscillations of the surface of units are arranged/located on the circumferences (parameters of gas change

along the radius), while with the tangential ones - at the diameters (parameters of gas charge along the circumference). Oscillations of both forms have the greatest amplitude in the cross sections near the head; in the measure approximation/approach for cross section at the nozzle entry their amplitude is decreased.

Radial oscillations more frequently appear in the large combustion chambers (with large relation $d_к/l_к$).

In the process of work ZhRD frequently simultaneously are present several modes of longitudinal, radial and tangential oscillations. The theory of acoustics establishes that with an increase in the frequency of oscillations increases/grows the scattered energy and, consequently, also the energy, necessary for maintaining the oscillations.

In connection with the conditions of chamber/camera ZhRD the smallest energy is required for maintaining the first modes of longitudinal and tangential oscillations and a somewhat high energy - for maintaining the first mode of radial oscillations. Therefore most easily are excited the first modes of longitudinal and tangential oscillations.

For the evaluation of the stability of chamber/camera to the

high-frequency oscillations frequently proves to be sufficient the account only of the first mode of longitudinal, radial and tangential oscillations.

Page 278.

The conditions of the emergence of the high-frequency oscillations, caused by the reaction of fluctuations of pressure with the processes in the combustion chamber, are determined by the relationship/ratio, analogous to equation (15.1):

$$\tau_{\Delta p_{\Phi} \to Q} = (1,\ 3,\ 5\ldots)\,\frac{T}{2} = (1,\ 3,\ 5\ldots)\,\frac{1}{2f}, \qquad (15.2)$$

where T and f - period and the frequency of the first mode of acoustic channel

VCh-oscillations/vibrations are maintained (or they are strengthened), if fluctuations of pressure p_K are found in the resonance with the fluctuations of the liberation of heat when the combustion zone is arranged/located in the region with the greatest change of the pressure, i.e., in the region of antinode.

If we, for example, expand one way or another combustion zone along the length of combustion chamber, then the liberation of heat in a whole series of cross sections, in the first place, in the cross

section where are located units, not only it will not maintain the longitudinal oscillations of pressure $p_к$ but on the contrary, it will depress them, damp.

Time $\tau_{\Delta p_\Phi - Q}$ has a basic effect on intrachamber instability, and the effect of time $\tau_{p_к - \Delta p_\Phi}$ and $\tau_{Q - p_к}$ can be disregarded/neglected.

Mechanism examined above of the maintenance of high-frequency oscillations makes it possible to base the following conditions for their supression:

1) the exception/elimination of the resonance of the oscillations of pressure $p_к$ and liberation of the heat:

2) the stretching of the process of burning in the time and the space.

The resonance indicated is eliminated by a change in time $\tau_{\Delta p_\Phi - Q}$ and significant dimensions and combustion chamber configuration. Period $\tau_{\Delta p_\Phi - Q}$ it is possible to affect, changing the quality of the atomization of propellant components, introducing in them these or other additives, etc.

The stretching of the process of burning in the time (i.e. a

change in time $\tau_{\Delta p_{\Phi}-Q}$ for the different parts of the propellant component flow) and in the space (with time $\tau_{\Delta p_{\Phi}-Q} \to \text{const}$ for all composite/compound component parts of the propellant component flow) is provided by construction/design and parameters of head, including by utilization of the swirl injectors with the different angles of the spray cone, of jet injectors with different range, by simultaneous utilization of jet and swirl injectors or mono- and duplex-fuel nozzles, etc.

The head of chamber/camera, as it was shown into §12.3, has the decisive effect on the stability of the process of fuel combustion.

Page 279.

High-frequency oscillations in the chamber/camera it is possible to suppress, selecting following structural/design data and parameters of the engine:

1) type and the construction/design of injectors; in a number of cases it is possible to ensure the stability of the process of burning, using injectors with baffle plate (see §12.2) or with the fan atomization instead of the injectors with the circular atomization;

2) a number of injectors, the order of their arrangement/position on the plane of head, and also the displacement of the ends/faces of the injectors of one propellant component relative to fire bottom (in particular, their submerging in the bottom);

3) an injector pressure drop; usually with its increase the amplitude of fluctuations of pressure p_k it is decreased;

4) the relationship/ratio of injector pressure drops of oxidizer and fuel, and also velocities of their injection into the combustion chamber; for example, for oxygen-hydrogen ZhRD the pressure differential on the hydrogen injectors and injection velocity of hydrogen it must be considerably more than for oxygen (injection velocity - 10 times and more).

The increased reliability possess the chambers/cameras, the supply of components into which is produced both into the precombustion chambers and it is direct into the combustion chamber (chamber/camera with the two-stage combustion). The process of burning in such chambers/cameras proceeds most completely and it is decreased the danger of the emergence of unsteady combustion, including with the lowered/reduced expenditures, since the dominant role in the fragmentation and the mixing perform the combustion

products, which enter from the precombustion chambers.

The possibility of the emergence of high-frequency oscillations (as VCh- oscillations/vibrations) depends on engine power rating: expenditure $\dot{m}$ and connected with it pressure p_K and coefficient $\varkappa$. In the period of the starting/launching when an injector pressure drop is still low, engine works less stably than in the nominal rating. If we increase pressure p_K by reduction in area $f_{кр}$, then the stability of the work of engine is raised.

The stability of the work of engine depends substantially on the type of propellant components; by of their correct trial and error and in certain cases by the addition of special additives (in a quantity to 1o/o) it is possible to avoid high-frequency oscillations.

The stability of burning affects the temperature of components of fuel/propellant, with which they are introduced into the chamber/camera; with its reduction/descent the burning becomes less stable.

In a number of cases the stability of burning in the chamber/camera is achieved by the setting up of the so-called acoustic partitions/baffles on the fire bottom within the combustion chamber perpendicular to it (Fig. 15.3).

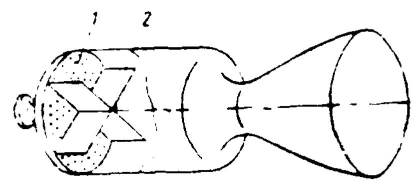


Fig. 15.3. Chamber/camera with the acoustic partitions/baffles in the area of fire bottom; 1 - fire bottom; 2 - acoustic partitions/baffles.

Page 230.

Such partitions/baffles, demarcating the cavity of combustion chamber near the fire bottom by several various volumes, decrease the tangential oscillations. Furthermore, partitions/baffles increase rigidity and strength of chamber/camera; they can be both cooled and uncooled burning. If for the stable chamber operation is sufficient the presence of partitions/baffles only with the starting/launching (tangential oscillations frequently appear precisely during this period), then it is expedient to use the burning partitions/baffles.

For damping the acoustic oscillations it is possible to use the acoustic damper- perforated/punched cylinder which is established within the combustion chamber, so that between the cylinder and its wall is formed the resonance cavity, which can be tuned for the

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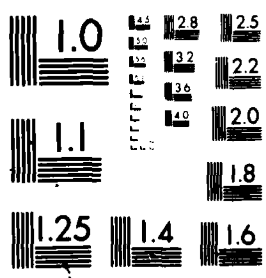

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specific expected frequencies of acoustic oscillations. Oscillations are damped as a result of the overflowing of combustion products through the openings/apertures and the interference of the straight/direct and reflected waves.

§15.2. Directions of perfection ZhRD.

Let us examine the bases of direction, on which is conducted the improvement of ZhRD.

An increase in the specific impulse ZhRD is achieved first of all by the selection of most effective fuels/propellants. Such fuels/propellants include O_2+H_2, F_2+H_2 and other fluorine-bearing fuels/propellants, and also fuels/propellants three-component and with the fuel containing metal (see §10.9).

Promising is the utilization of a mixture of supercooled liquid and solid hydrogen ("sludge").

Specific impulse ZhRD is raised also upon transfer to the more advanced diagrams (diagram "gas- liquid" and etc.).

An increase in the stability of work of ZhRD is provided with the aid of the measures, examined in §15.1.

The decrease of the dimensions of engine can be achieved/reached by transition/transfer to higher pressures $p_{\text{к}}$, rational layout of engine, by improvement of the construction/design of its aggregates, in particular, by the utilization of chambers/cameras with the inner body and so forth; the use/application of the latter becomes possible in connection with the creation of new structural materials, thermo-insulating coatings and development of the effective methods of cooling the walls.

Is very promising the engine of large thrust, which consists of the large quantity (24 and more) of combustion chambers, located on the ring around the common nozzle with the inner body.

For reducing/descending the mass of engine they are used:

1) structural materials with the high specific strength (titanium, aluminum, magnesium, beryllium and their alloys, plastic and the combined materials);

2) advanced technology of the production: vacuum casting, welding by electron beam, electric spark and electrochemical treatment, treatment by lasers, diffusion welding so forth;

3) the intensification of the processes, which take place in the engine accessories, in this case is decreased not only the mass of engine, but also its dimensions and volume.

Page 231.

One of the trends of development of ZhRD is an increase in their thrust. Utilization of ZhRD with the large thrust in the engine installation of the first booster stages significantly simplifies it, since is decreased a necessary quantity of engines and is facilitated the task of their simultaneous starting/launching.

In a number of cases for expanding the possibilities of application of in series made ZhRD appears the need in an increase in their thrust without any structural/design alterations or with their smallest quantity (first of all can be required changes in construction/design TNA and ZhGG), for which usually is increased the propellant component flow with the appropriate increase in pressure p_K. ZhRD it is possible to boost/force on the thrust also by the addition of high-energy component, for example by the addition of fluorine into oxygen. The important direction of perfection ZhRD are the safeguard of simplicity and convenience in the operation, and

also the safeguard of readiness of engine for the immediate launching/starting.

For expanding the region of the possible use/application of engine decisive importance have its following special features/peculiarities:

1) large reliability;

2) the possibility of multiplying, including in the vacuum;

3) the possibility of large changes in expenditure of $\dot{m}$, pressure p_k and coefficient $\varkappa$ and, therefore, change in the thrust over wide limits, with the retention/preservation/maintaining of sustained combustion and high value φ_β.

4) the reserves for the development of construction/design, in particular increase in the service life of work;

5) efficiency of afterward different (among other things prolonged ones) time intervals between the inclusions/connections;

6) creation by the engine of control forces and moments/torques;

7) the admissibility of prolonged storage in that charged/filled of states;

8) the adaptability of engine to changes in the environmental parameters. From this point of view are effective

a) the engines whose chamber/camera is equipped with the mobile section of nozzle; b) the engines, which have nozzle with the inner body, and c) the engines with the high-altitude nozzle, controlled and work at the level of sea.

Reduction of cost/value and the time of the development of engines and their production is the most important problem, especially for the first-stage engines of rockets.

Page 282.

Chapter XVI.

SOLID-PROPELLANT ROCKET ENGINES.

§ 16.1. Device RDTT [solid-propellant rocket engine].

The simplest diagram and the principle of rocket thrust-chamber firing of solid fuel are examined in § 1.2. The characteristic feature of RDTT is the fact that its housing is simultaneously combustion chamber and peculiar tank for positioning/arranging the fuel charge, in connection with which drops off the necessity for the propellant feed system.

RDTT consists of housing with the nozzle, fuel charge and aggregates of different systems (ignition, the creation of control forces, disconnection, etc.).

Housing most frequently has a form of cylinder, but it can be

spherical. On the front/leading bottom of housing in the center usually is placed torch igniter. Back plate is frequently performed together with the nozzle.

Essential effect on the construction/design of RDTT exert the special feature/peculiarity of the arrangement/position of fuel charge in the housing.

Page 283.

Are distinguished the deposit and fastened fuel charges. Deposit charges are manufactured separately and they insert in the housing in the form of one or several cylinder grain. The case-bonded charges usually prepare via drenching fuels/propellants in the liquid state directly into engine block. After hardening this charge provides strong/durable cohesion/coupling with the walls of housing and possesses certain mechanical strength. The usually bonded charges have central channel.

Fig. 16.1 and 16.2 depict diagrams of RDTT with deposit and case-bonded charges. Deposite charges are fixed by the end/face of rear bottom and by the resilient spacer in front/leading bottom, which makes possible the thermal expansion of charge both during the storage and with the work of RDTT. Deposit charge can lean also on

the diaphragm with the openings/apertures, adjusted on the nozzle entry. The case-bonded charges are connected with the walls of housing through the layer, which receives the loads, caused by different temperature expansion of charge and housing with the storage and the work of RDTT. Furthermore, the layer indicated shields the walls of housing from the thermal loads.

For the contraction the lengths of RDTT use the so-called countersunk nozzle (Fig. 16.3), which, besides the fact that foreshortened length of RDTT, possesses the series/number of the advantages in front of the usual nozzle, namely:

1) are decreased mass and cost/value of engine;

2) is decreased necessary effort/force for deviating the nozzle (for the system of the creation of the controlling efforts for rocket vehicle);

3) are improved working conditions of the coupling assembly of the unit of housing with the nozzle: in the area of the unit indicated is formed the stagnation zone of combustion products, so that heat fluxes in it are decreased.

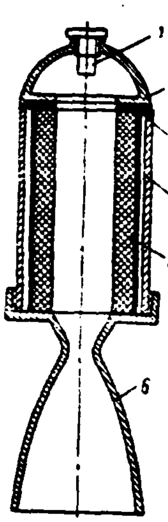
Nozzle can be submersed into the housings to 40-80o/o, and in

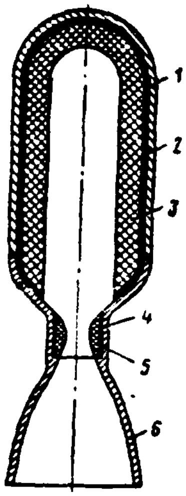
certain cases and completely. However, the greater the value of the immersion of the nozzle (it it is possible to characterize ratio l/l_k see Fig. 16.3), the greater degree to which is decreased specific impulse.

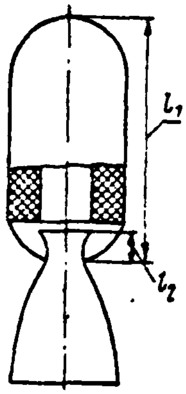
For increasing the length of RDTT instead of one nozzle can be used, especially for RDTT with high pressure p_k, several nozzles (Fig. 16.4). However, in this case due to the additional losses specific impulse RDTT also descends.

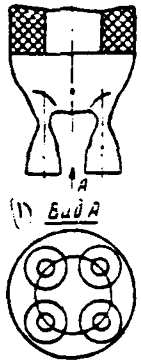
In RDTT with the deposit charge housing they connect with the nozzle with the aid of the flanges or the thread (see Fig. 16.2) while in RDTT with the case-bonded charge - the welding (see Fig. 16.1). Housing can be prepared together with the nozzle, for example via the coil/winding of tape from the fiberglass to the mount/mandrel of the required profile/airfoil.

The axes/axles of housing and nozzle must coincide precisely for elimination of eccentricity of thrust. In certain cases according to the designs of RDTT in the composition of the flight vehicle of nozzle it can be inclined at an angle to the axis/axle of housing to 45° (Fig. 16.5).









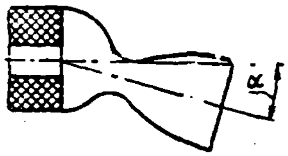


Fig. 16.1. Fig. 16.2. Fig. 16.3. Fig. 16.4. Fig. 16.5.

Fig. 16.1. RDTT with depcsit charge: 1 - pinwheel igniter; 2 - front/leading bottom; 3 - ply; 4 - wall of housing; 5 - depcsit charge with armcring coating; 6 - nczzle.

Fig. 16.2. RDTT with case-bonded charge: 1 - wall of housing; 2, 4 - layer of thermal insulaticn; 3 - case-bonded charge; 5 - insert/bushing from high-temperature (strength) material; 6 - nozzle.

Fig. 16.3. RDTT with countersunk nczzle.

Fig. 16.4. Housing of RDTT with fcur nozzles.

Key: (1). Form.

Fig. 16.5. RDTT with nozzle, established/installed at angle α to axis/axle of housing.

Page 285.

For example, this deflection of nozzle for RDTT of the rockets, hung to the wings of aircraft, creates thrust in the direction of the lift of wings and eliminates the effect of combustion products on them.

§ 16.2. Requirements for RDTT.

RDTT is chemical rocket engine. Therefore for RDTT by analogy with ZhRD must be provided the high values of the set of the parameters β, specific pulse $I_{уд}$ and density of fuel/propellant $\varrho_т$.

However, to RDTT is imposed a number of specific requirements, moreover the main thing among them is the specific (prescribed/assigned) and stable law of rate of combustion. The law of rate of combustion is called a change in rate of combustion of fuel charge in the operating time of engine (depending on pressure $p_к$ and other parameters).

Rate of combustion of fuel charge is equal to the distance, passed in 1 s by flame front perpendicular to burning surface. The speed indicated they desigrate by letter U and they express in mm/s. Therefore the law of rate of combustion is dependence U=f(τ).

The law of rate of combustion determines together with [illegible] parameters of charge the mass flow rate per second of combustion products. The surface area, over which burns fuel charge, is called burning area and they designate by letter S. At rate of combustion U and burning area S the volume of the fuel/propellant, which burned down 1 s, is equal to product US. With the multiplication of the product indicated by the density of fuel/propellant $\varrho_т$ we obtain the formula of the mass flow rate per second of combustion products

$$\dot{m}=US\varrho_т. \quad (16.1)$$

Consequently, the mass flow rate per second of combustion products $\dot{m}$ is determined by speed and burning area, and also by density of fuel/propellant.

Flow rate $\dot{m}$ determines the mode of operation of RDTT, namely pressure in chamber/camera $p_к$ and thrust in vacuum $P_п$. Actually/really, on the basis of equations (4.14) and (5.10)

$$p_к=\frac{\dot{m}\beta}{f_{кр}}; \quad (16.2)$$

$$P_п=\dot{m}\beta K_p. \quad (16.3)$$

If we provide prescribed/assigned the law of rate of combustion, then parameters $p_к$ and $p_п$ at the work of RDTT in the vacuum are constant. These parameters at the conditions indicated are called nominal.

Page 288.

Deviations $p_к$ and $p_п$ from the nominal values make characteristics worse RDTT and rocket vehicle as a whole on the following reasons.

1. Housing RDTT rely on strength on the basis of nominal pressure of products combustion $p_{к.ном}$. If the actual pressure of combustion products $p_{к.д}$ to the noticeable degree exceeds value $p_{к.ном}$, under condition $p_{к.д}>p_{к.ном}$ can occur the destruction of housing RDTT.

Under condition $p_{к.д}<p_{к.ном}$ the characteristics of RDTT deteriorate, since:

a) increase/grow losses to the dissociation;

b) is not used the available safety factor of housing; if condition $p_{к.д}<p_{к.ном}$ occurs during entire operating time of RDTT, this then means that it would be possible to attenuate of wall and

mass of housing, and this in the case in question is not done.

2. Thrust of RDTT, as any other type of rocket engine, is determined flight conditions of rocket vehicle. If in the process of work of RDTT its reaction force to the noticeable degree differs from nominal value, then the creation of the reliable system of flight control of rocket vehicle hinders.

Rate of combustion of solid fuel in the process of work of RDTT must to least possible degree to differ from the prescribed/assigned law.

Rate of combustion of solid fuel affect in essence the following factors: a) pressure $p_к$, b) the initial temperature of charge $T_{нач}$; c) the composition of solid fuel and d) the speed of the motion of products along the burning surface.

The dependence of rate of combustion of solid fuel on pressure $p_к$ can be different, but most frequently this dependence they depict in the following form:

$$U_{p,T}=U_{p_{нсх},T_{нсх}}p_к^{\nu}, \qquad (16.4)$$

where $U_{p,T}$ - rate of combustion at the arbitrary values of the pressure of the products of combustion and temperature of solid fuel;

$U_{p_{нсх}}, T_{нсх}$ - rate of combustion at the initial pressure of combustion products (for example, with 1 bar) and at a datum temperature of fuel/propellant;

ν— the constant index, which depends only on the type of solid fuel.

Coefficient ν is determined experimentally. Its value is within the limits from 0.2 to 0.8 (sometimes to 0.85). The analysis of equation (16.4) shows that with the decrease of coefficient ν the dependence of rate of combustion on pressure $p_к$ is decreased.

Page 287.

When $\nu>0{,}85$ the solid fuel is very sensitive even to the low deviations in technology of the manufacture of fuel charge and therefore it is dangerous in the inversion. In the case $\nu>0{,}85$ a small change of the surface of burning leads to a sharp increase in pressure $p_к$, which, in turn, increases rate of combustion of fuel/propellant.

The temperature effect of solid fuel on the rate of its burning

is estimated at the invariable pressure by the relationship/ratio

$$u_{p_{нсх},T} = u_{p_{нсх},T_{нсх}} \frac{B}{B-(T-T_{нсх})},$$

where B - temperature combustion rate coefficient of solid fuel. For many fuels/propellants it is equal to 350-450.

But as a result of the low thermal conductivity of solid fuel the temperature of fuel charge with the work of RDTT remains the same as it was before its inclusion/connection. Therefore rate of combustion depends on the initial temperature of fuel charge.

With a decrease in the initial temperature of fuel/propellant the rate of its burning is decreased. In this case in accordance with equation (16.1) is decreased the flow rate of combustion products $\dot{m}$, and consequently, pressure $p_к$ and the thrust of RDTT. The period of combustion of fuel charge (and, consequently, the operating time of engine) is increased, but total impulse is decreased only to the low degree (due to the decrease of thrust coefficient with decompression $p_к$).

With an increase in the initial temperature of fuel/propellant rate of combustion, pressure $p_к$ and thrust of RDTT are increased, and the period of combustion of fuel charge is decreased; the total impulse increases to a small degree (due to an increase in the thrust

coefficient with an increase in pressure p_{κ}).

In order to decrease the effect of ambient temperature on rate of combustion of solid fuel, for RDTT is created the so-called microclimate. For example, rockets of RDTT can be stored and employed in the warmed (thermostatically controlled) containers. Thermostatic control makes it possible to maintain the temperatures of solid fuel in the narrow range, which provides the stability of the engine characteristics. Variations of the temperature of RDTT during the arrangement/position of rockets in the launching silo, and also in the starting/launching well of the submarine it is substantially less than on the open starter.

The spread of characteristics of RDTT at different initial temperature of fuel charge can be decreased by change of area by replacing of nozzle (in the small RDTT) or insert/bushing in the critical cross section.

The rate of combustion of solid fuel depends on the rate of the motion of combustion products along the burning surface; this motion occurs, for example, for the fuel charges with the internal duct, the rate of combustion products in proportion to approach toward the end of the channel of charge increasing/growing.

Page 288.

The dependence indicated can be written in the form

$$\frac{U_W}{U}=f(W),$$

where U - rate of combustion, designed from formula (16.4);

U_W - rate of combustion in the presence of the motion of combustion products along the burning surface with a rate of W.

Rate of combustion of solid fuel affect also other factors, including: a) technology of the manufacture of fuel charge (size of the particles of the fuel/propellant, the quality of their mixing, the compacting pressure, etc.);

b) the linear acceleration of rocket vehicle (due to the emergence of stresses/voltages in the discharge and the effect of acceleration on the particles, which are located on the burning surface);

c) special additives in the fuel charge, the increasing or gearing down burning (see § 16.3).

It is possible to ensure the wide range of rate of combustion of the solid fuels: from 0.25 to 250 mm/s. Rate of combustion of the

most frequently used solid fuels is from 1 to 50 mm/s.

The parameters of RDTT are mutually connected with each other: a change in one parameter leads to a change in another, and the latter can, in turn, change the first parameter, etc., this dependence can cause the significant deviation of the real mode of operation of RDTT from the nominal. For example, an accidental increase in flow rate of $\dot{m}$ calls an increase in pressure p_K; in this case it increases/grows flow rate $\dot{m}$, which, in turn, leads to increase p_K and so forth. Another example to this dependence: an increase in the initial temperature of solid fuel causes an increase in values of U and m, in this case increases/grows pressure p_K and consequently also U, etc.

Besides those indicated above, to the solid fuels and the fuel charges of RDTT are imposed also the following requirements.

1. Low low pressure limit. Low pressure limit is called such pressure p_K lower than which the burning of solid fuel becomes unstable. This limit depends on the type of solid fuel.

2. Good combustibility and stability of burning.

3. Reproducibility of characteristics of fuel charge in series production RDTT.

4. Physicochemical stability of fuel charge. It is characterized by the resistance of fuel charge to the initiation of cracks, volatilization of the separate composite/compound component parts of the fuel/propellant, absorbtion of moisture and aging in the process of prolonged (of up to several [illegible] storage, including under conditions of oscillating the ambient temperature, for example, from 213°K [-60°C] to 333°K [+60°C].

Page 289.

Solid fuel is most sensitively to crack initiation with they are low temperatures. The presence of cracks can cause a sharp increase in the burning area and pressure $p_к$, which can lead to the decomposition of engine.

Solid fuel must possess the smallest hygroscopicity, i.e., it must not absorb and hold moisture. Otherwise considerably become complicated the manufacture of charges and operation of RDTT. In order to exclude the effect of moisture on the fuel charge with storage of engine, the cavity of the chamber/camera, in which it is placed, it is hermetically sealed with the aid of the silencer/plug. It is installed usually in nozzle throat is thrown out by the

pressure of combustion products during engine starting.

Aging fuel charge during the storage is explained by the course of the slow spontaneous processes of decomposition and is characterized by a change in the structure, by the disturbance of continuity and by a deterioration in the strength characteristics of fuel charge; therefore the noticeable aging of fuel charge is inadmissible for RDTT.

5. Sufficient simplicity and safety of technology of production in fuel charge, transport and storage of RDTT. In solid fuel as much as possible there should not enter dangerously explosive and toxic substances. It must be low-sensitivity to the effect of friction and impacts/shocks.

§ 16.3. Composition and the properties of fuels/propellants of RDTT.

By the chemical composition the solid rocket propellants (TRT) subdivide into the colloidal ones and the mixture ones.

Colloidal solid rocket propellants are the solid solutions of the uniform homogeneous substances whose molecules contain combustible and oxidative elements/cells. Most frequently as the colloidal solid rocket propellant is used the solid solution in which

it is contained by 50-75o/o of nitrocellulose and 43-25o/o of nitroglycerin. With an increase in the content of intro-glycerin in the solid rocket propellants is increased the specific impulse of RDTT, but deteriorate the mechanical properties of charge, is raised explosiveness and deteriorates the physicochemical stability of fuel/propellant.

Charges of the colloidal solid rocket propellants provide the sufficiently high values of specific impulse, possess good mechanical properties and low low pressure limit of 5-10 bars [~5-10 kgf/cm²]. Their shortcomings include:

a) by fire and explosiveness during the manufacture of fuel charge;

b) the increased values of index v;

c) the insufficient elasticity, which eliminates the possibility of the fastening of charge with the walls of housing.

Page 290.

Due to the noted shortcomings the charges of the colloidal fuels/propellants are used in essence in comparatively small ones of

engine.

Mixture solid rocket propellants, being mechanical mixtures of oxidizers and fuels, call also heterogenic (heterogeneous). The oxidizers of mixture solid rocket propellants are the inorganic crystalline substances: in essence is used ammonium perchlorate NH_4CLO_4: more rarely are used potassium perchlorate $KClO_4$, sodium nitrate $NaNO_3$, of nitrate of potassium KNO_3 and nitrate of ammonium NH_4NO_3. Therefore for creation of fuel charge with required mechanical strength one of the combustible mixture solid rocket propellants must possess the connecting/cementing properties: a comparatively small quantity of fuel must be sufficiently in order to connect into the strong/durable charge a large (to 88o/o) quantity of crystal oxidizer and second combustible mixture solid rocket propellant (powder-like metal). In this case the composition of solid rocket propellant to the greater degree approaches stoichiometric, which increases the specific impulse of RDTT.

As the fuel-bonds of mixture solid rocket propellants serve the synthetic polymeric organic compounds: polysulfide, polyurethane and polybutadiene natural rubbers, etc. As the second fuel the solid fuels which is intended for an increase in their energy properties, use in essence powder-like aluminum, less frequent - beryllium and magnesium.

In the contemporary mixture solid rocket propellants it is usually contained by 60-75o/o of oxidizer, 15-25o/o of fuel-bond and 10-20o/o of aluminum.

To a number of advantages of the mixture solid rocket propellants over colloidal ones they relate:

a) greater specific impulse;

b) very low low pressure limit;

c) the wider permissible range of initial temperatures of charge;

d) the high density of fuel/propellant;

e) large thermal stability and more prolonged permissible period of storage;

f) the ease of fabrication of fuel charge and RDTT (possibility of drenching directly into engine block).

Into the composition of colloidal and mixture solid rocket propellants usually are introduced the additives, which make with the possible:

a) pressing charge, delay/retarding/deceleration or the acceleration of hardening charge (for the mixture fuels/propellants) and so forth (technological additives);

b) prolonged storage of charge (stabilizers);

c) an increase to silt a decrease in the velocity of burning (catalysts and the stabilizers of the speed of combustion), etc.

Composition and characteristics of some solid rocket propellants are given in Table 16.1.

Page 291.

Table 16.1. Characteristics of some solid rocket propellants [2].

(1) Тип топлива		(2) Коллоидные		(3) Смесевые	
(4) Марка топлива		JPN (США)	Н (СССР)	—	—
(5) Состав топлива в % по массе		(6) Нитроцеллюлоза — 51,5 (9) Нитроглицерин—43,0 (10) Диэтилфталат—3,25 (12) Присадки—2,25	(6) Нитроцеллюлоза—57,0 (9) Нитроглицерин—28,0 (11) Динитротолуол—11,0 Стабилизатор и технологические присадки—4,0 (13)	(7) NH_4ClO_4 Полиуретан Алюминий	(8) NH_4ClO_4 Полибутадиен Алюминий
ϱ	(14) $кг/м^3$	1610	--	1680	1700
(15) U_0 при $p_к$=70 *бар* и температуре 293° K (16)	(17) *мм/сек*	16,5	7,3	5,6	12,0
ν	—	0,69	0,60	0,05	0,24
$T_к$	°K	3200	2350	3600	3300
(15) $I_{уд}$ (при $\varepsilon_c=20\div100$)	(18) *н·сек* / (19) *кг*	1960—2255	1960—2205	2600	2600
	(19) *кГ·сек* / (19) *кг*	200—230	200—225	265	265
(20) Допустимые значения $p_к$	(21) *бар*	>35	>40	>1	>1
Допустимые значения $T_{зар}$	°K	243—333	243—323	—	—

Key: (1). Type of fuel/propellant. (2). Colloidal. (3). Mixture. (4).

Brand of fuel/propellant. (5). Propellant composition in o/o by mass. (6). Nitrocellulose. (7). Polyurethane aluminum. (8). Polybutadiene aluminum. (9). Nitroglycerin. (10). Diethylphthalate. (11). Dinitrotoluene. (12). Additives. (13). Stabilizer and technological additives. (14). kg. (15). with. (16). and to temperature. (17). m/s. (18). N•s. (19). kg•s. (20). Allowed values.

FOOTNOTE [1]. $T_{зар}$ - temperature of charge solid rocket propellant. ENDFOOTNOTE.

Page 292.

§ 16.4. Solid-propellant grain.

The construction/design of solid-propellant grain must provide:

a) filling of housing of RDTT with fuel;

b) exception/elimination (as far as possible) the contact of the hot combustion products with the walls of housing for facilitating the working conditions for their;

c) the invariability of the geometric form of charge with the loads, which appear with the transport of RDTT, and also with its

work when on charge act axial (tc 15-120 g_3) and lateral (1-7.5 g_3) g-forces, intensive vibration in the wide frequency spectrum and the pressure of hot combustion products;

d) the smallest remainders of the unused fuel (toward the end of the work of engine).

The geometric form of solid-propellant grain is determined by required change-in-thrust pattern by time and by technology of the manufacture of charge.

On the basis of equations (16.1) and (16.3) the thrust of RDTT for this type of solid fuel under condition p_K=const is directly proportional to burning area.

Depending on character charges in the burning area during the work of engine distinguish the following types of charges (Fig. 16.6):

a) with the constant burning area (with the neutral burning);

b) with the increased burning area (with the progressive burning);

c) with the decreased area of combustion (with the degressive burning);

d) with the burning area, which is changed according to prescribed law.

Fig. 16.7 depicts the graphs of a change in pressure $p_к$ in the operating time of engine during the neutral, degressive and progressive burning. The same charges of solid propellent can be subdivided into the geometric special features/peculiarities of burning surface into the following types:

a) charges with the endburning (see Fig. 16.6a);

b) charges with the comprehensive burning (see Fig. 16.6b);

c) charges with internal burning (see Fig. 16.6c).

In charges with endburning the time of combustion is determined by the length of charge, while in charges with the internal burning - web thickness (the wall thickness of the cylinder of charge). The surface of the charge on which must not be burning, shield by the armoring coating (inhibitor). Charge is wrapped with tape from inhibitor or glued with sheets of inhibitor with the aid of the glue.

To preferably use charges whose inhibitor does not wash by the products of combustion, in particular case-bonded charges whose as inhibitor serves a layer of thermal insulation on the internal surface of the wall of housing.

Page 293.

Charges with end burning are used in essence for RDTT with low thrust, the long operating time and the fuel/propellant with the low energy characteristics. Shortcomings in such charges they are:

a) the large dry mass of housing; its walls undergo intensive heating with the work of engine; therefore it is necessary to increase their thickness for the safeguard of sufficient strength;

b) the displacement of the center of mass of engine with the burning of charge.

Charges with the comprehensive burning are used comparatively rarely.

The greatest advantages possess the case-bonded charges with the internal burning. They burn over the surface of internal duct; therefore hot combustion products do not come into contact with the

walls of housing. Solid fuel, as already mentioned, possesses the very low coefficient of thermal conductivity. Therefore the case-bonded charge protects the walls of housing from the effect of heat fluxes, which makes it possible to create prolongedly operating RDTT without the external flowing cooling.

Furthermore, the fastened charges give the following advantages in comparison with the inserted.

1. More effectively is used volume of housing, since there are no clearances between charge and wall of housing and are not required additional supports for charge (it is held directly by walls of housing).

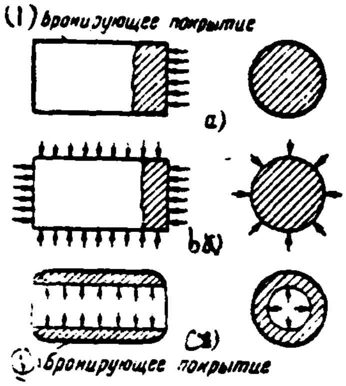


Fig. 16.6.

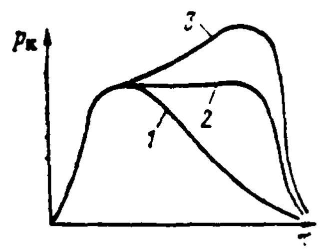


Fig. 16.7.

Fig. 16.6. Types of fuel charges: a) with constant burning area (with endburning); b) with increased burning area (with comprehensive burning); c) with decreased burning area (with internal burning).

Key: (1). Armoring coating.

Fig. 16.7. Graphs of change of pressure of products of combustion in combustion chamber in time for charges with degressive (1), neutral (2) and progressive (3) burning.

Page 294.

2. Are increased rigidity and strength of thin-walled engine block, since charge supports housing during transport and other

operations/processes with rocket, and during flight it is able to absorb some mechanical loads, caused by effect/action of linear accelerations and pressure of combustion products. Therefore during the utilization of the case-bonded charges the necessary wall thickness is decreased (in the case of applying high-strength steel - to 1-3 mm), which decreases the mass of RDTT.

3. Is simplified technology of manufacture of solid-propellant grain (especially large-size charges).

4. Is decreased possibility of appearance of cracks, since case-bonded charges, prepared directly with drenching into housing of RDTT, are polymerized at low temperatures and possess plastic properties.

The example of the case-bonded charges are the poured charges with the internal duct in the form of the 7-ray star (Fig. 16.8), which ensure constant burning area. Their certain shortcoming is the fact that the end of the burning there are formed the pieces in volume 3-5o/o of the total volume of charge which either do not burn or they burn out at the reduced pressure, which leads to the decrease of specific impulse of RDTT.

Fig. 16.8 depicts also other charges with the internal burning: a) with channel in the form of 5-ray star and b) with four

longitudinal slots; they are usually turned to the side of nozzle; the presence of such slits provides the required law of burning.

Between the case-bonded charge and the wall of housing is placed a layer of thermal insulation. Between a layer of thermal insulation and a charge it can be placed layer of adhesive cluster (for example, synthetic resin), which fastens them between themselves.

Charge, layer of thermal insulation, layer of adhesive cluster and wall of housing must be reliable connect/joined together. The presence of laminations on the boundaries (for example, due to the oscillation/vibration of temperature at the prolonged storage) can lead to the emergency of engine in the process of its work.

Difficulties presents the drenching of the case-bonded charges without the formation of air cavities in them; as crack, they are not admitted, since they lead to an increase in the burning area, which can cause the decomposition of engine.

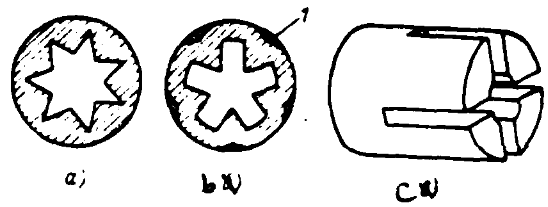


Fig. 16.8. The types of the fuel charges: a) with the internal duct in the form of 7-ray star; b) with the 5-ray star 1 with the rays/beams of slot form; c) charge with the internal duct and four longitudinal slots on the part of the length of charge; 1 - insert from the inert material (foam plastic).

Page 295.

For simplification in technology of manufacture, control/check, transport, storage of large-size charges and reduction in the cost/value of the manufacture of large/coarse RDTT are used sectional fuel charges.

Sectional charge (Fig. 16.9) consists of several consecutive parts (sections) of housing with the fastened fuel charges, which have internal duct.

RDTT they assemble from the sections on the starting or intermediate positions of section they connect between themselves

with the aid of the detachable (for example, flanged) or permanent (welded) joint. Welded joint provides the best airtightness and smaller mass of RDTT, than flanged, but are produced the specific inconveniences with the assembly of RDTT on the launching site.

The advantage of sectional charges consists also of the fact that, having one standard type of sections, it is possible, using their different quantity, to obtain these or other values of thrust and operating time of engine (i.e. to change the total impulse of thrust of RDTT over wide limits).

In order to decrease the difficulties, which appear during the manufacture of large fuel charges, and to exclude stress concentration in the sharp angles of charges, are used segmental fuel charges (Fig. 16.10). They consist of several segments which are divided from each other on the surfaces of contact with the layer, an employee by the simultaneously armoring coating and by the resilient spacer.

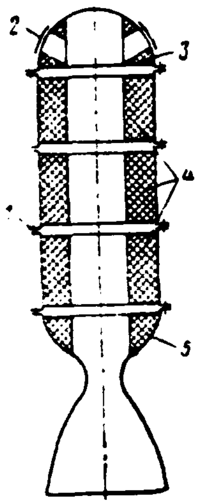


Fig. 16.9.

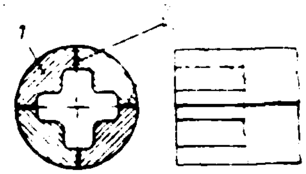


Fig. 16.10.

Fig. 16.9. RDTT with sectional charge: 1 - flange joint; 2 - cover/cap of nozzle of counterthrust with explosive device; 3 - section of charge with upper bottom, 4 - middle sections of charge with sections of housing; 5 - section of charge with back plate and nozzle.

Fig. 16.10. Segmental fuel charge: 1 - segmental grain: 2 - armoring coating.

Page 296.

With the constant section of the internal duct of fuel charge

the rate of combustion products in proportion to approximation/approach to a nozzle, as already was indicated, increases, which can lead to the erosion of canal surface. For the exception/elimination of the phenomenon indicated the channel can be performed by conical, with the expansion to the nozzle.

For a considerable (to 5 times and more) increase in rate of combustion of solid fuel it is possible to use a uniform arrangement/position of short heat conducting wires from copper, aluminum, silver, magnesium or tungsten on the volume of fuel charge. During the arrangement/position of the wires indicated perpendicular to burning surface the rate of combustion is approximately/exemplarily two times more than during their parallel arrangement/position.

Into the fuel charge it is possible to build in long longitudinal wires and different wire constructions, in this case is increased not only rate of combustion of solid fuel, but also the mechanical strength of fuel charge. The utilization of metals in the solid rocket propellants complicates the resolution of the problem of the protection of the walls of engine from the increased heat fluxes and the errosive effect of products of combustion, which contain the condensed particles.

The determination of geometric dimensions and form of charge of RDTT precedes the thermodynamic calculation which in many respects is analogous to the corresponding calculation of ZhRD, but it is noticeably more complex it; this is explained by the more complex composition of the products of combustion of charge of RDTT, which certain moreover, condensed phase.

As a result of thermodynamic calculation is determined the specific impulse of RDTT in vacuum $I_{уд.п}$, after which from prescribed/assigned value $P_п$ it is possible to find fuel consumption per second

$$\dot{m}=\frac{P_п}{I_{уд.п}}.$$

The total necessary quantity of fuel/propellant $m_т$ is equal

$$m_т=\dot{m}\tau,$$

where τ - prescribed/assigned operating time of engine.

Most simply is designed the turning area of end type fuel charge (it is constant during entire operating time of RDTT)

and its length

$$S=\frac{\dot{m}}{\varrho_т U}$$

$$l_{зар}=\frac{m_т}{S\varrho_т}.$$

Page 297.

For the fuel charge with the internal duct it is necessary to calculate its initial form and area of cross section $f_{кан}$, and also the length of charge $l_{зар}$, which determine primary surface of the burning

$$S=f_{кан}l_{зар}.$$

Web thickness of charge, which ensures the prescribed/assigned operating time of engine,

$$e_{св}=U\tau.$$

Greatest difficulty presents calculation of rate of combustion U (see § 16.2), especially if in the process of burning area S and, therefore, values $\dot{m}$ and $p_к$ change.

With volume and mass of fuel charge are connected the following characteristics of RDTT.

1. Solidity/loading factor of RDTT, equal to ratio of volume of fuel charge $V_{зар}$ to combustion chamber volume $V_к$,

$$K_V=\frac{V_{зар}}{V_к}.$$

This coefficient depends on form and type of charge and is equal to:

a) 0.8-0.97 - for the charges with the endburning;

b) 0.8 - for the deposit charges with the internal burning;

c) 0/9 - for the case-bonded charges with the internal burning.

2. Coefficient of mass of RDTT, which is ratio of mass of fuel/propellant to mass of engine:

$$K_m = \frac{m_т}{m_{дв}}.$$

The coefficient of mass for RDTT with the deposit charge is equal to approximately/exemplarily 0.75, and for RDTT with case-bonded charge -0.8 - 0.95. Value K_m increases/grows during the utilization of materials with the high specific strength. With an increase in mass coefficient of RDTT simultaneously increases/grows relation $m_т/m_{нач}$ of rocket vehicle (rocket step/stage)and, consequently, also its characteristic velocity.

§ 16.5. Comparative characteristic RDTT and ZhRD.

The type of engine (RDTT or ZhRD) for each concrete rocket vehicle can be selected on the characteristic velocity, in this case must be taken into consideration the special features/peculiarities of both RDTT and ZhRD.

To a number of basic special features/peculiarities of RDTT due to which in a number of cases by it is given up the preference over

ZhRD, relate the following.

1. Simpler construction. However, it is necessary to note that in proportion to perfection of RDTT it gradually becomes complicated.

Page 234.

2. Smaller cost/value and shorter time of development, especially large/coarse RDTT. With increase in the complexity and transition/transfer to the more effective solid fuels/propellants the advantage of RDTT indicated over ZhRD becomes less noticeable.

3. Large reliability, explained, first of all, by simplicity of construction/design.

The engine installation, which consists of several RDTT, to more simply create, and it to a certain extent more reliable than the installation, which consists of several ZhRD, since each solid propellant engine in DU is independent variable.

4. High density of solid fuel, especially in comparison with liquid propellants in which as fuels serves liquid hydrogen. Usually the starting mass of step/stage with RDTT more than with ZhRD, but its overall dimensions is less than for the step/stage with ZhRD. The high density of solid fuel partly compensates main disadvantage of

RDTT in comparison with ZhRD - relatively low specific impulse.

5. Simpler starting/launching, especially under low-pressure conditions and all the more so in vacuum (in outer space), and so under conditions of weightlessness. In particular, for the starting/launching of RDTT under the conditions indicated is not required the creations of artificial acceleration.

6. Large simplicity and safety of operation. For the launching of rocket RDTT is required a less complex launcher and a smaller quantity of service personnel, than for the launching of rockets with ZhRD. On the surface and submerged boats of rocket RDTT to employ more safely than rocket with ZhRD. The explosiveness (among other things explosiveness from the detonation) of many solid fuels is relatively low, while for the series/number of liquid propellant components it significant. The toxicity of solid fuels weak, and many liquid propellant components are toxic. During the operation of rockets RDTT in contrast to the rockets with ZhRD does not appear the problems, connected with the vaporization of propellant components.

7. Relative simplicity of storage, including under conditions of outer space. Solid fuel barely exerts effect on the metals, while cryogenic liquid propellant components add to metals brittleness, but the series/number of liquid propellant components causes the severe

corrosion of many metals.

Nevertheless the wide application of RDTT is significantly impeded by their following special features/peculiarities (in comparison with ZhRD).

1. Noticeably smaller specific impulse. However, in proportion to the development of more effective solid fuels the specific impulse of RDTT increases/grows.

2. Substantially smaller possibilities of changing in thrust and repeated or multiplying of engine.

Page 299.

This special feature/peculiarity of RDTT is explained, in particular, by the great difficulty of changing the flow rate per second of the combustion products of solid fuel. Furthermore, with a reduction/descent in their combustion pressure, as has already been indicated, becomes unstable, but at low pressures solid fuel can not at all burn.

3. Smaller resource/lifetime of work. Usually it does not exceed 150 s, especially for RDTT of high thrust, that use high-energy solid

fuels, due to the difficulties of the protection of nozzle liners from the heat fluxes.

4. Dependence of characteristics on ambient conditions, especially on ambient temperature, which determines temperature of fuel charge. Therefore the basic parameters of RDTT (pressure p_k and thrust) can be secured with the the large spread of values in comparison with ZhRD (see § 16.2). The temperature range in which RDTT reliably works, is less than for ZhRD. In ZhRD the ambient temperature affects in essence the density of liquid propellant components, but this effect can be comparatively easily considered.

5. High cost of solid fuel (in comparison with usual liquid propellants: by liquid oxygen, kerosene, etc.). For manufacturing the fuel charges (special of large ones) are required powerful/thick plants. It should be pointed out that the cost of solid fuels has a tendency toward the reduction/descent, and high-energy liquid propellants (fluorine, hydrogen, etc.) are sufficiently expensive and almost inaccessible.

6. Difficulty of transport of completely equipped rockets of RDTT (especially large rockets). However, is possible servicing rockets of RDTT with solid fuel on the spot of start (for space and other used for the peaceful purposes rockets) or comparitively not

far from the places of utilization of RDTT (on the intermediate bases).

§ 16.6. Special features/peculiarities of the protection of walls RDTT from the heat fluxes.

The nozzle liners and bottom of the housing frequently are protected from the heat fluxes with the aid of the ablation cooling, and in the area of the throat install nozzle insert/bushing of graphite. Greater thermal and erosive resistance possess the inserts/bushings, prepared from the high-temperature (strength) materials: molybdenum, tungsten, etc.

Effective is the following laminated construction/design of nozzle liners of RDTT (Fig. 16.11). Metallic power wall 6 is covered with from within several layers from different materials. The greatest thickness has a layer of pyrolytic graphite (pyrographite) 3, which possesses the significant anisotropy of the properties: thermal conductivity in one direction is close to the thermal conductivity of copper; in the perpendicular direction the pyrographite is virtually non-heat-conductive.

Page 300.

This property of it makes it possible to abstract/remove heat fluxes from the area of critical cross section to the less heat-stressed sections of nozzle. A layer of pyrographite is the base layer, which absorbs heat fluxes.

To the pyrographite will be brought in thin thermoresistant layer 1 of the tungsten which simultaneously shields pyrographite from the errosive effect of combustion products. Between a layer of tungsten and the pyrographite is placed interlayer 2, which prevents the diffusion of carbon into a layer of tungsten and reduction/descent in its mechanical properties.

A metallic wall and a layer of pyrographite are divided between themselves with a layer of plastic 5 and with a layer of ceramic insulation 4. The vaporization of plastic and other physicochemical processes in it absorb certain part of the heat fluxes. Ceramic insulation decreases the heat fluxes into a layer of plastic.

§ 16.7. Special features/peculiarities of systems RDTT.

Ignition system of fuel charge. For inflaming the fuel charge it is necessary to warm thoroughly surface layer to a temperature at which begin the processes of decomposition, inflammation and combustion of fuel/propellant, and to also create in the combustion

chamber the pressure, at which fuel charge stable burns. Ignition system must provide the sufficiently short and stably reproducible ignition delay of fuel charge (usually 5-45 ms) and eliminate overshoot of the pressure of combustion products during engine starting.

In RDTT is used the pyrotechnic, pyrogenic and hypergolic ignition of fuel charge. The elements of ignition system are assembled in the front/leading bottom or they install in the silencer/plug of nozzle.

Cartridge ignition is produced by the pinwheel igniter which operates/wears from electric fuse with the bridge of incandescence or with the exploding wire bridge.

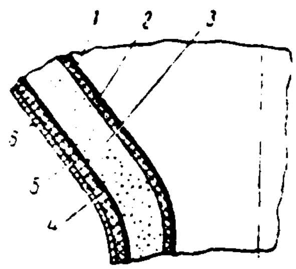


Fig. [illegible]. The laminated construction/design of nozzle liner of RDTT: 1 - thermoresistant layer of tungsten; 2 - interlayer; 3 - layer of pyrographite; 4 - layer of ceramic thermal insulation; 5 - layer of plastic; 6 - external nozzle liner.

Page 301.

In the system of pyrogenic ignition is used ignition of TGG, which is actually the solid-propellant rocket engine of relatively low sizes/dimensions. The hypergolic ignition of fuel charge is accomplished/realized by supplying the starting/launching liquid or gaseous oxidizer (ClF_3, BrF_5, etc.), which ignites spontaneously with the contact with the grain surface.

System of a change in the thrust. Before examining the methods of changing the thrust, let us note that even retention/preservation/maintaining of the invariable mode of operation of RDTT presents difficulties in connection with the

dependence of rate of combustion of solid rocket propellant on pressure $p_к$ (see § 16.2).

The invariable mode cf cperaticn of RDTT can be maintained crly, if with the probable deviaticns cf pressura $p_к$ from the ncminal value, as a result of affecting/acting one or the other factors, pressure $p_к$ again is reduced, i.e., $p_{к.д} \rightarrow p_{к.ном}$, that it is possible to achieve, if $\nu<1$. Curve 2 (Fig. 16.12) characterizes the flcw cf the gases, which are generated with ccmbusticn of solid rocket propellant ($\dot{m}_2$), while curve 1 - flcw cf the gases, which escape behind nozzle ($\dot{m}_1$). The invariable mcde of cperation of RDTT is provided under condition $\dot{m}_1=\dot{m}_2$ morecver flow rate $\dot{m}_2$ it is determined according to equation (16.1), and flcw rate $\dot{m}_1$ - according to equation (16.2).

With $\dot{m}_1>\dot{m}_2$ or $\dot{m}_2<\dot{m}_1$ is disturbed the state cf the equilibrium between the flow cf the gases, which are generated in the combustion chamber and which escape from it. For example, with an accidental increase in pressure $p_к(p_{к.д}>p_{к.ном})$, as can be seen frcm Fig. 16.12, will cccur relaticnship/ratic $\dot{m}_2<\dot{m}_1$, which will lead to reduction $p_к(p_{к.д} \rightarrow p_{к.ном})$.

Analogcus ones by consideraticns it is pcssible to show that under conditicn $\nu>1$ the invariability of the mode cf operaticn of RDTT becomes impossible (see Fig. 16.12).

A change in the thrust or RDTT in the flowing signals of the system of control of rocket vehicle to a considerable degree is hindered/hampered (see § 16.4). Therefore use the previously selective (programmed) stepped or smooth change the thrusts (it is necessary to consider the spread, specific RDTT, see [illegible]).

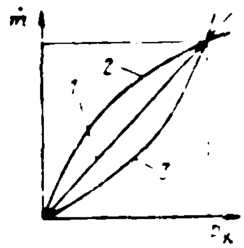


Fig. 16.12. Graphs $\dot{m}=f(p_K)$ 1 - flow rate of the combustion products, which escape behind the nozzle; 2 - flow rate of the combustion products, which are generated with combustion, that are generated with the combustion of solid rocket propellant with index ν [illegible] 3 - flow rate of the combustion products, which are generated with combustion of solid rocket propellant with index ν [illegible].

Page 302.

Thrust can be changed, selecting one or the other geometric form of the fuel charge (see § 16.4) or using composite/compound charges, i.e., the charges whose individual parts (layers) are prepared from the fuels/propellants with different rate of combustion. For example, in the composite/compound charge with the internal burning it is possible internal and skins to prepare from the fuels/propellants with different speed of burning, which gives the excess of thrust of RDTT in the starting mode/conditions five-ten times in comparison with the thrust under operating conditions.

System of the creation of control forces. For the creation of control forces in RDTT in essence use rotary or diverged nozzles, and also liquid injection (N_2O_4, etc.) into the expanding section of nozzle.

System of disconnection. In the simplest case of RDTT it does not have the system of disconnection, in this case the engine works until burns down almost entire fuel charge. Disconnection of RDTT at the required moment of time is achieved by a sharp increase in the area of the openings/apertures, through which occurs the outflow of combustion products, i.e., critical cross section. For this purpose are opened/disclosed the additional nozzles (are jettisoned their covers/caps, for example with the aid of the blasting cord) or basic nozzle is disengaged housing.

With a sharp increase in the area of critical cross section are decreased the pressure of combustion products and heat fluxes from them to the burning surface, as a result of which are disturbed the conditions of the course of the reaction of burning.

If additional nozzles are placed on the front/leading bottom, then during their opening is created counterthrust, which can be used for the department/separation of the mastered rocket step/stage from the upper stages; engine it will not produce further increase in the

rate of rocket, in spite of the fact that it still completely was not turned off (it continues to create certain thrust). For this it is necessary simultaneously with the delivery of the command to the engine cutoff to tear its mechanical bonds with the upper stage or the nose section.

In order to decrease the remainders of the unused fuel and to reduce the time of drop of thrust of RDTT with the charge, which has channel in the form of star with the fact or another number of rays/beams, are used the rods of the inert material with the low density (for example, foam plastic) which place on the periphery of the charge (see Fig. 16.8b).

§ 16.8. Stability of the mode of operation RDTT.

During the adjustment of solid-propellant rocket engine can be revealed the instability of the operating mode, which leads to its decomposition or to the inadmissibly large vibrations of engine and rocket vehicle as a whole.

Depending on whether does correspond the frequency of pressure p_{K} to the acoustic modes of the combustion chamber volume or not, is distinguished acoustic and nonacoustical instability [10].

Page 303.

Nonacoustical instability is characterized by the fact that its frequencies are considerably lower than the acoustic modes. The instability indicated is observed when the ratio of the free (not occupied with fuel charge) combustion chamber volume $V_{к\,св}$ to the throat area $f_{кр}$ is less than certain critical value. Nonacoustical instability is observed in essence in the small RDTT; it is possible to avoid it, respectively increasing relation $V_{к\,св}/f_{кр}$.

Acoustic instability is explained by the presence of the time between moments of the effect of pressure wave on the grain surface of solid rocket propellant and by the moment/torque of an increase in the flow rate of combustion products (delay).

The source of the energy, which feeds oscillations of the volume of combustion products in the combustion chamber, is the heat, which is isolated in the process of burning of solid rocket propellant.

Acoustic instability can appear both when the external (with respect to the process of burning) impulse/momentum/pulse of disturbance/perturbation is present and in its absence (intrachamber instability).

An example of acoustic instability with the external impulse/momentum/pulse of disturbance/perturbation is the longitudinal instability, caused by abrupt change in pressure $p_{к}$. This change can appear, for example, at the moment of the passage of the separated part of the heat shield or igniter through the critical cross section (with the throw-out of igniter behind the nozzle). The instability indicated is observed in essence in the large ratios of the length of the free combustion chamber volume $l_{к.св}$ to its diameter $d_{к.св}$, namely in relation $l_{к.св}/d_{к.св}>10$ and for the mixture of solid rocket propellants, which contain in their composition aluminum.

Any measure, directed toward the exception/elimination of the possibility of an abrupt change in pressure $p_{к}$ or on its decrease, contributes to an increase in the stability of RDTT. The same target it is possible to achieve by decreasing of pressure $p_{к}$ or change in the composition of solid rocket propellant, which leads to an increase in the rate of its burning.

Intrachamber acoustic instability is characterized by a gradual increase in the amplitude of fluctuations of pressure $p_{к}$ from very low ones to the inadmissibly high values, moreover in essence are observed purely the tangential or longitudinal oscillations of the first mode. This instability is produced by synchronism of fluctuations of pressure $p_{к}$ with the oscillations/vibrations of the

flow rate of combustion products and is determined by the character of processes in the combustion zone and by the time of the course of the corresponding reactions.

For the exception/elimination of acoustic instability is necessary the excess of the losses of the acoustic energy above its influx, called by the process of burning.

The losses of acoustic energy are determined by damping (absorbing energy) properties of the products of combustion and free combustion chamber volume.

Page 304.

The damping properties possess only the products of the combustion of the solid rocket propellants containing metal, having in their composition condensed particles.

All rigid surfaces, with which contact the combustion products (surface of fuel charge, the surface of front/leading and back plate) also possess the damping properties to a certain degree. The properties of the surface of fuel charge indicated are improved with an increase in its elasticity. The same purpose serves the setting up of antiresonant rings or disks with the openings/apertures of one or

the other form in both bottoms, and also the setting up of antiresonant porous rods in the central channel of charge; in them occurs the absorption of acoustic energy.

Taking measures indicated above, including changing the combustion chamber configuration and composition of solid rocket propellant, it is possible to raise the stability of the mode of operation of RDTT.

§ 16.9. Main trends of perfection of RDTT.

Effectiveness of RDTT to the high degree depends on the type of solid fuel. Therefore special attention is paid to the creation of high-energy solid fuels.

Contemporary solid rocket propellants, on foreign specialists' views, possess comparatively low rate of combustion, and their characteristics to a strong degree depend on ambient temperature. By selecting the composition of solid rocket propellant and technology of its manufacture they attempt to obtain high-energy fuels/propellants with a good reproducibility of mechanical properties, the lowered/reduced sensitivity to the temperature and the moisture, by resistivity to aging, by the wide speed range of burning and by the increased allowed values of axial and transverse

accelerations.

Analogous with fuels/propellants of ZhRD the promising oxidizers of mixture solid rocket propellants are the fluorine-bearing compounds. A number of high-energy fuels includes the fuels into molecule of which instead of carbon is introduced boron or aluminum.

The decrease of the mass of construction/design of RDTT is achieved by reduction of pressure p_{K}, during the utilization of the case-bonded charges with the internal burning, by applying new technology of the manufacture of housing and nozzle (for example, by coating of metal on the fuel charge electrolytically), and also by the use/application of materials with the high specific strength: titanate and aluminum alloys and glass-fiber-reinforced plastics. To the promising materials for the housings of RDTT relate the glass-fiber-reinforced plastics, reinforced by fibers on the basis of boron, tungsten, carbon and beryllium.

For changing the thrust of RDTT without an essential deterioration in its characteristics it is possible to use the following methods.

Page 305.

1. Change in throat area by displacing inner body along axis/axle of nozzle or by supplying working medium/propellant (liquid or gas) into critical cross section through annular slot or belt/zone of openings/apertures. Gas for this purpose can be selected/taken from the combustion chamber. The utilization of the methods indicated hinders by the severe conditions for the work of inner body and by the relatively low value of a change in the thrust in the case of the input/introduction of working medium/propellant into the critical cross section.

2. Input/introduction of chemically active liquid (F_2, CF_2, ClF_3, ClF_5, N_2O_4, etc.), which ignites spontaneously with solid fuel. Changing fluid flow rate, it is possible to change the engine thrust in the relation from 20:1 to 50:1. Strictly speaking, this engine is no longer solid propellant engine, but occupies the intermediate position between RDTT and RDGT, differing from the latter in terms of the fact that the liquid (liquid component) does not exert a substantial influence on rate of combustion of solid rocket propellant; furthermore, the engines indicated are distinguished by the ratio of the flow rate of liquid component to the flow rate of the combustion products of fuel charge; it for RDTT with the liquid component is equal to 0.5-0.6 and for RDGT - 2-5. In RDTT with liquid component is simplified the problem of the cooling: using external flowing cooling, it is possible to use high-energy solid rocket

propellants and fuel charges with the endburning.

3. Change in rate of combustion of solid rocket propellant by changing in its temperature or blowout of burning surface by additional gas. The temperatures of charge it is possible to change via the transmission of cold liquid or hot combustion products through the tubes, uniformly arranged/located in the charge.

4. Increase in burning area of charge in any forced manner, for example by decomposition of armoring coatings charge by chemically active liquid component.

5. Utilization of compensating charges. If we in the housing of RDTT, besides the main fuel charge, place several additional (compensating) charges, then at low pressure $p_к$, caused by a reduced temperature of the main charge, it is possible to include the compensating charges thereby to decrease the spread of characteristics of RDTT with the work under different temperature conditions.

For an increase in the operating time of RDTT during the utilization of high-energy solid rocket propellants it is possible to use internal (among other things porous) cooling, moreover by coolant can it serves liquid (ammonia, hydrocarbon fuel, etc.) or gas (for

example, the combustion products of the charge of special TGG).

Effectively porous cooling with the use/application of boiling metals (Li, Na, Mg, Zn, Cu, Ag, B) or their hydrides (LiH, etc.) as the coolant.

Page 306.

For the realization of this cooling the nozzle all over length or on the subcritical part is supplied with two walls, moreover internal wall is performed from the porous high-temperature (strength) metal, for example tungsten. Between the walls is placed the coolant. With the work of RDTT the coolant, receiving the heat fluxes entering, is heated, is melted and vaporizes; the forming vapors pass through porous diaphragm into the flow of combustion products. Porous cooling can be used and only for nozzle lining.

For repeated connection and disconnection of RDTT is necessary the system of the repeated inflammation of charge and its extinguishing. For the repeated inflammation of fuel charge it is possible to use a hypergolic ignition system, while for the repeated damping - supply of special working medium/propellant (liquid, gas or the sublimated powder) to the burning surface of charge. Working the body indicated must not impede engine restart. As liquid for the

damping of charge can serve water and aqueous solutions of alkali salts (latter have lower freezing point).

One of the directions in the development of solid-propellant rocket engines is an increase in their thrust to the very high values - to 40 MN (~~4000 T). Are tested RDTT with the diameter of housing 6.6 m and thrust 16 MN (~~1600 T), the duration of tests reaching 135 s.

A reduction/descent in the cost of RDTT is achieved by the method of simplification in technology of the manufacture of charges, research of the ways of reduction in the cost/value of solid fuels and by the repeated utilization of housings and nozzles of RDTT.

Page 307.

Part I7.

NONCHEMICAL ROCKET ENGINES.

To the nonchemical rocket engines together with thermal type ERD, electrostatic and electromagnetic belong as is indicated in §1.6 YaRD, SRD and gas RD [1].

FOOTNOTE [1]. Chapters XVII and XVIII are written by whole based on materials of foreign sources [5, 21, 49, 50, 51, 52]. ENDFOOTNOTE.

Besides cold and hot gas engines, can be used the engines, working medium/propellant of which is the superheated water, which is located in the chamber/camera under the large pressure and which is converted during the motion along the nozzle into the vapor. For the work of such engines just as for cold gas RD, there is no necessity to have a source of heat on board the rocket vehicle. For the work other nonchemical thermal RD (YaRD, SRD and thermal type ERD) this

source necessary (see §1.6).

The heat, applied to the working medium/propellant in YaRD, SRD and thermal type ERD, produces different processes (melting, vaporization, sublimation, ionization, etc.), as a result of which are formed gaseous products or plasma; their outflow behind the nozzle creates thrust.

Design parameters and the type of working medium/propellants in all types RD select from the condition of obtaining the maximum characteristic velocity of rocket vehicle. Therefore it is necessary to consider not only specific impulse RD, but also mass of DU; in particular, it is important so that the source or the receiver of primary energy of engine installation would possess large specific output power, i.e., by the output power, which falls on 1 kg. of their mass (mass of source or the receiver of primary energy).

Important characteristic of DU with nonchemical thermal RD is their efficiency η, which is the ratio of the power of the stream of working medium/propellant $N_{\text{стр}}$ to the thermal source power of energy $N_{\text{тепл}}$, i.e.

$$\eta=\frac{N_{\text{стр}}}{N_{\text{тепл}}},$$

or during the replacement of absolute powers by specific ones (on 1 n or on 1 kgf thrust of RD)

$$\eta=\frac{N_{\text{стр.уд}}}{N_{\text{тепл.уд}}}. \qquad (17.1)$$

Page 308.

With an increase in the efficiency of DU is decreased the necessary thermal of the source power of energy for the creation of one and the same thrust, and consequently, is decreased its mass.

As working medium/propellants nonchemical thermal RD can serve the liquid substances (H_2, NH_3, N_2H_4, CH_4, C_3H_8, etc.), liquified inert gases (He, Ar, N_2, etc.), liquid metals and their hydrides (in particular, Li and LiH), the sublimated substances (for example, camphor), etc. Gas engines work on nitrogen, helium, oxygen, argon, hydrogen and other gases. In the rocket engines are piloted spacecraft it is possible to use as the working medium/propellants products cosmonauts' vital activities (in particular, CO_2).

Chapter XVII.

Nuclear rocket engines.

In the chemical reactions participate only the electron shells of atoms, and their nuclei remain invariable, by which are determined relatively low quantities of heat, which is isolated in the presence of the chemical reactions, and the low values of specific impulse ZhRD. Substantially higher specific impulse can be obtained, if to use fission reactions or nuclear fusion special of the substances (see §1.6). For example, for fission reaction of uranium the heat, due to 1 kg. of fissionable material (taking into account "mass defect"), $3.7 \cdot 10^5$ once is more than a quantity of heat, which is isolated in the presence of the most powerful/thickest chemical reaction on 1 kg. of fuel/propellant.

In heat exchange YaRD the heat, which is isolated as a result of nuclear fission, is used for heating of working medium/propellant. Behind the nozzle of such engines flow out the pairs of working medium/propellant. Fissionable material it remains in the chamber/camera, and its mass decreases only as a result of nuclear decomposition. Heat can be transmitted from the fissionable material to the working medium/propellant by convection and by

radiation/emission.

Exhaust jet of mixture YaRD is the mixture of decay products and working medium/propellant.

Nuclear fuel/propellant in YaRD can be solid (monolithic or powder-like), liquid or gaseous.

Of all types YaRD to the greatest degree are finished heat exchange engines with solid radioisotopes (radioisotope RD) and YaRD with the reactor, in which is accomplished/realized the division of solid nuclear fuel/propellant.

As working medium/propellant YaRD with the reactor divisions can serve liquid, and radioisotope - liquid and solids.

Page 309.

§17.1. Special features/peculiarities of requirements for the working medium/propellant YaRD.

For obtaining the large specific impulse YaRD it is necessary to ensure the high enthalpy of working medium/propellant at the nozzle entry. Therefore in accordance with equation (4.20) at a selected

maximum permissible temperature, which does not call the fusion of fissionable material (2500-3000°K), most effective is working the body, which possesses large heat capacity c_p (taking into account the effect of the reactions of dissociation). Value c_p to a certain degree is determined by molecular weight μ. Therefore, other conditions being equal, in YaRD to more preferably use working a body with the low value of molecular weight, primarily hydrogen. At high temperatures, characteristic for YaRD, the molecular hydrogen dissociates to the atomic. With the high content of atomic hydrogen in the composition of working medium/propellant its molecular weight approaches a lowest possible value - to the atomic mass of hydrogen (μ_H=1,008). Therefore specific impulse YaRD with the solid fissionable material can reach 8000-8500 N•s/kg [~-800-850 kg•s/kg], in spite of the fact that the temperature of working medium/propellant in such engines lower than temperature of combustion products chemical RD.

The dissociation of working medium/propellant not only decreases its molecular weight, but also absorbs heat, what for YaRD is positive factor (see §3.3), since in this case is simplified the problem of its cooling. As for any thermal RD, in this case it is desirable so that the process of expanding the working medium/propellant in the nozzle would be equilibrium.

Cooling is one of the most complex problems, which appear during

creation of nuclear engines. Therefore working body of YaRD must possess good cooling properties; furthermore, to avoid the inadmissible superheating of any parts nuclear engine must be very thoroughly designed as heat exchanger.

One of the best working medium/propellants of YaRD is liquid hydrogen. Heat capacity c_p of gaseous hydrogen at temperatures, characteristic for YaRD, is 5-10 times more than in the combustion products of usual chemical fuels/propellants, which in essence determines the high specific impulse of hydrogen YaRD.

As working medium/propellant of YaRD can serve also other substances, in this case the most important requirements, presented to them, are high density and ability easily to dissociate at high temperatures. Such liquid substances include hydrazoic fuels (NH_3, N_2H_4 and etc.), the water, alcohols and hydrocarbons (CH_4, C_3H_8, etc.). They provide smaller specific impulse YaRD, than liquid hydrogen, but they have substantially high density and are possible the cases, when their use/application proves to be more effective.

§17.2. Radioisotope engines.

Simplest type nuclear RD are radioisotope engines.

Page 310.

Within the chamber/camera of radioisotope engine (Fig. 17.1) is placed the capsule with the radioisotope, moreover between the chamber wall and the capsule is radial clearance into which enters working the body from the tank. Capsule is prepared from the high-temperature (strength) metals, in particular made of the tungsten.

Basic requirements for the radioisotopes, utilized in the engines, they are:

a) the greater possible quantity of heat, which is isolated on 1 kg. of the mass of radioisotope;

b) the sufficiently large period of the half-life (time, during which decays exactly the half atomic nuclei the substances); the half-life period designate $\tau_{1/2}$;

c) the insignificant level of γ-radiation (for reducing/descending the mass of biological protection).

In proportion to the nuclear decomposition of radioisotope the quantity of heat, isolated data by the mass of radioisotope,

continuously is decreased. However, when selecting of radioisotope with the corresponding period of half period the specific impulse of engine, although it descends, remains within the permissible limits.

The most completely indicated above requirements satisfy the radioisotopes of polonium (Po^{210}) and plutonium (Pu^{238}) (numerals indicate the atomic mass of isotope). One kilogram of mass Po^{210} and Pu^{238} allots $5 \cdot 10^8$ and $6.1 \cdot 10^8$ the kilo-joule of heat respectively. Period of the half-life for Po^{210} —————— is 0.378 years, and for Pu^{238} - 86.8 years.

Radioisotope is placed into the capsule not in the pure form, but in the connection with other substances, which makes it possible to raise the temperature of its melting and in the required degree to lower a quantity of isolatable heat in order not to allow/assume the fusion both the isotope and capsule.

By the special feature/peculiarity of radioisotope engines is the impossibility of control decomposition/decay of radioisotope that it limits the possibility of changing in thrust and specific impulse of the engines indicated.

Radioisotope engines it is most expedient to use for the creation of low thrusts. DU with such engines possess smaller

specific mass, simpler by the construction/design can be developed in the shorter time than DU with the electrical rocket engines, but they are inferior to the latter in the value of specific impulse.

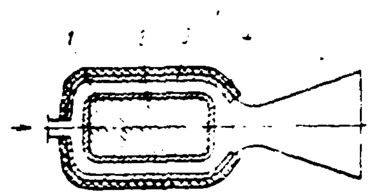


Fig. 17.1. Chamber/camera radioisotope of RD: 1 - shell of the capsule: 2 - radioisotope: 3 - chamber wall; 4 - thermal insulation.

Page 311.

§17.3. Processes in the fission-type nuclear reactor.

In the rocket engine with the fission-type nuclear reactor must flow/occur/last the controlled chain reaction of nuclear decomposition. More easily are split the atomic nuclei of elements/cells with the large atomic mass, since with an increase in the atomic nuclear mass they become less stable. To a number of natural elements with the greatest atomic mass relate uranium and thorium.

As fissionable materials for YaRD can serve U^{235}, Pu^{239} and U^{233} the period of half-life of which is equal to $8.8 \cdot 10^8$; $2.6 \cdot 10^4$ and $1.6 \cdot 10^5$ years respectively. Most widely as the fissionable material they use U^{235}.

In natural uranium it is contained by 99.82o/o of isotope U^{238} and 0.712o/o of isotope U^{235}. Pu^{239} is formed from the natural isotope U^{238} by the effect on it of neutrons, and isotope U^{233} - thus from natural thorium 232 (Th^{232}) [28].

Nuclei U^{235} split via their neutron bombardment. The incidence/impingement of neutron into the nucleus causes its artificial excitation, in this case the energy of nucleus is increased so, that it becomes unstable and decays with the formation of new neutrons, nuclei with the smaller mass and the liberation of a large quantity of heat. If after the nuclear fission, caused by neutron capture, at least one of the newly neutrons being produced causes the decomposition of new nucleus, then nuclear reaction will maintain itself; therefore this reaction is called chain.

Chain reaction in nuclear reactor YaRD must be controlled: the power of nuclear reactor, determined by a number of nuclear fissions per unit time, must yield to control according to predetermined program. The uncontrollable chain reaction, which is accompanied by a sharp increase of a number of nuclear fissions per unit time, leads to explosion of reactor.

Depending on the energy level of neutrons, the generatrices during the nuclear fission, subdivide into the rapid ones, the intermediate ones and the slow ones (thermal). Slow neutrons possess the smallest energy, compared with the kinetic energy of the molecules of gas. Such neutrons provide the nuclear fission of isotope U^{235}.

Since during the nuclear fission of isotope U^{235} are isolated fast neutrons, and for maintaining the chain reaction are required slow neutrons, is necessary the delay/retarding/deceleration of fast neutrons. For this purpose on their way place the special retarders, as which can serve the substances, which little absorb neutrons. The possibility of a decrease in the velocity of neutrons makes it possible to control/guide chain reaction. The additional advantage of the utilization of a retarder is an increase in the surface, through which the heat is transmitted from the fissionable material to the working medium/propellant. ¶ Page 312.

As retarder can serve substances and elements/cells with a low atomic mass of: graphite, heavy water, beryllium, oxide of beryllium, etc. The less the atomic mass of element/cell, the more rapidly slow down the neutrons during the motion in its medium.

The quantity of fissionable material, necessary for heating of

working medium/propellant to the necessary temperature, is very small. The quantity of fissionable material, which must be placed in the reactor, is selected from other consideration, in essence from the condition of obtaining the critical mass, and also from the condition of sufficient surface area, through which the heat is transmitted from the fissionable material to the working medium/propellant.

Critical mass is called the least quantity of fissionable material, in which is possible the maintenance of chain reaction. For the pure/clean isotope U^{235} the critical mass composes sphere by radius into several ten millimeters and mass of 20-30 kg. During the utilization of a retarder the critical mass increases/grows to hundreds of kilograms.

Due to the limitations, connected with the critical mass, it cannot be created YaRD of low thrust with the reactor, in contrast to they are radioisotope engines.

§17.4. Device and the specific parameters of solid-phase reactors.

Fission-type nuclear reactors are distinguished by the type of active region/core. Active region/core is called the part of nuclear reactor in which is located the fissionable material. Active

region/core can be solid-phase, liquid phase and gas-phase. This division is conventional, although extensively it is used.

Let us examine the device of nuclear reactor with the solid-phase active region/core on the thermal neutrons. This reactor is placed within the engine chamber (Fig. 17.2). Its active region/core is several rods of the retarder (for example, graphite), in which evenly distributed the fissionable material.

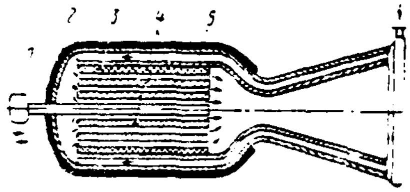


Fig. 17.2. Chamber/camera YaRD with the solid-phase reactor: 1 - control rod with the drive: 2 - protective shield; 3 - reflector; 4 - fuel elements; 5 - coolant passage of chamber/camera.

Page 313.

For achievement of the high temperature in the reactor core it is necessary to use fissionable materials with possibly the higher melting point: the dioxide of uranium UO_2 ($T_{пл}=3075° K$), uranium-tungsten and uranium-zirconium alloys, the solid solution of carbide of uranium and carbide of niobium UC-NbC ($T_{пл}=3500° K$) and etc. During the utilization of the latter the temperature of active region/core it is possible to raise to 3025-3075°K and to heat working body to 2775°K.

Around the reactor core is placed the reflector, which is the shield, which returns the neutrons, which left the active region/core, conversely. Consequently, reflector contributes to the retention/preservation/maintaining neutrons and decreasing the critical mass of fissionable material. For the reflector are used the

same materials, as for the retarder.

The heat, liberated by rods (fuel elements), is transmitted to the working medium/propellant, which takes place in the space between the rods.

Nuclear reactor must be equipped by the control system of the rate chain fission reaction of nuclei, in which they are included:

a) the sensors, which measure the neutron flux density in the reactor;

b) the control rods which actively absorb neutrons;

c) the electromechanical devices, which ensure a change of the submersion depth of the rods indicated into the active region/core.

Density of neutron flux in the active region/core determines a number of nuclear fissions per unit time and, consequently, also the power of reactor.

The controlling/guiding and emergency rods are prepared from the substances which to a considerable degree absorb neutrons. A number of such substances includes cadmium, boron and carbide of boron B_4C.

If neutron density in the active region/core on any reason begins to be increased, then on the signal of the sensor, mounted in the active region/core, is given command for the greater immersion of control rods into the active region/core.

In this case the neutrons are absorbed also to the greater degree, which leads to a reduction/descent in the power of reactor up to the prescribed/assigned level.

Emergency rods are used in the system of the emergency disconnection of the reactor: in the case of spontaneous sharp growth of the power of reactor emergency rods sharply are introduced into the active region/core, providing rapid attenuation of chain reaction.

The outer covering of nuclear reactor is protective shield, which protects other parts of the engine and service personnel or crew from the nuclear radiation: γ-rays and fast neutrons. The absorption of γ-rays occurs only during the collision with the electrons. Therefore the shields, which shield from the γ-rays, it is expediant to prepare from the elements/cells with the large atomic mass (tungsten, bismuth, the connections of lead, etc.), the nuclei

of which contain a large quantity of electrons.

Page 314.

Neutrons best anything are seized by materials with the low atomic mass (B, Li, Be, etc.). Are sufficiently effective for the protection from the fast neutrons substances with the large density of the atoms of hydrogen in the molecule: NH_3, H_2O and liquid hydrogen. At high temperatures is most effective hydride of lithium LiH.

Thus, for the protection from the γ-rays and the fast neutrons it is necessary to make shields from a thick layer of element/cell with the large atomic mass and a layer of material with the low atomic mass.

During the utilization of graphite as the retarder basic problem is the destructive effect of the flow of hydrogen on graphite, which leads to corrosion and carry-off or its surface layers. In order to avoid this, to the surface of graphite will be brought in the protective coating (for example, a layer of carbide of niobium).

Solid-phase call not only reactors whose fissionable material is found in the solid state, but also reactors with the fuel elements,

under solid shell of which is placed fissionable material in the liquid state.

Diagram YaRD with the solid-phase reactor is examined into §1.2. In addition to Fig. 1.3 it is necessary to indicate that the temperature of the working medium/propellant, selected/taken behind the nozzle for the drive of turbine, must descend by means of displacement with the cold working medium/propellant. The power of turbine changes with choke/throttle with electric drive, adjusted on the feed line of turbine by working medium/propellant.

Let us determine the values of the specific powers YaRD based on the example of engine with the solid-phase reactor, moreover let us accept $I_{\text{уд.п}} = 8000$ N•s/kg [~~800 kg•s/kg] and $P_{\text{п}} = 900$ kn [~~90 T].

In accordance with equations (1.22) and (1.29) the specific power of exhaust jet of engine in the vacuum

$$N_{\text{с·р.уд.п}} = \frac{I_{\text{уд.п}}}{2} = 4 \overset{(1)}{\text{кВт/Н}} \left[\approx 40 \overset{(2)}{\text{кВт/кГ}}\right].$$

Key: (1). kW/N. (2). kW/kg.

If we take as for the work in the vacuum thermal efficiency η_{t}, equal to 0.8 and to disregard/neglect all other losses, then taking into account equation (17.1) the specific and absolute heat output of nuclear reactor and engine as a whole are equal to

$$N_{\text{тепл.уд.п}} = \frac{N_{\text{стр.уд.п}}}{\eta_t} = 5 \overset{(1)}{\text{квт/н}} \ [\approx 50 \overset{(2)}{\text{квт/кГ}}];$$
$$N_{\text{тепл.п}} = N_{\text{тепл.уд.п}} P_{\text{п}} = 4500 \overset{(3)}{\text{Мвт}}.$$

Key: (1). kW/N. (2). kW/kg. (3). MW.

Knowing the heat output of nuclear reactor, it is possible to determine the volume of active region/core and the mass of reactor. For this purpose are used the following specific parameters.

Page 315.

1. Specific volumetric heat output of reactor $N_{V_{\text{а.з}}}$, i.e. heat output, which falls per unit volume of reactor core $V_{\text{а.з}}$:

$$N_{V_{\text{а.з}}} = \frac{N_{\text{тепл}}}{V_{\text{а.з}}}.$$

Tentatively it is possible to accept $N_{V_{\text{а.з}}} \approx 10 \cdot 10^6$ kW/m³. As is evident, on 1 m³ of the volume of reactor core falls very large power.

2. Specific (on 1 m³ of active region/core) mass of reactor i.e.

$$m_{V_{\text{а.з}}} = \frac{m_{\text{яр}}}{V_{\text{а.з}}},$$

where $m_{\text{яр}}$ - mass of reactor.

In the approximate computations of the mass of reactors YaRD with the thrust in several hundred Kilonewtons [several ten ton-forces] it is possible to accept $m_{\Gamma_{a.3}} \approx 8$ $m/м^3$.

§17.5. Special features/peculiarities yard with the liquid phase and gas-phase reactor.

If we use a fissionable material not in the solid, but in the liquid or gaseous state, then it is possible to increase substantially the temperature of active region/core and working medium/propellant and to attain a significant increase in the specific jet firing (for YaRD with the gaseous reactor - to 2-5•10⁴ N•s/kg [~-2000-5000 kg•s/kg]).

In many types of liquid phase and gaseous reactors in contrast to the solid-phase ones working the body is mixed with the fissionable material. Heat from the fissionable material to the working medium/propellant is transmitted in the liquid phase reactors by the convection path and radiation/emission, and in the gas-phase reactors - in essence by radiation/emission. The departments/separations (separation) of working medium/propellant indicated from the fissionable material, moreover precisely the problem of the retention of fissionable material in the liquid phase ones, and especially in the gas-phase ones, the reactors presents

greatest difficulty. The throw-out of fissionable material together with the working medium/propellant inadmissible, since in this case:

a) descends the power of reactor and the engine thrust:

b) considerably is increased the cost/value of the engine (is more accurate the cost/value of its testing);

c) exhaust jet, which contains fissionable material, it causes radioactive contamination of the surrounding space.

The stability of the course of chain reaction in the liquid phase and gas-phase reactors is provided with the aid of the thick layer of a reflector-retarder, placed on the periphery of reactor.

Page 316.

One of the possible diagrams of YaRD with the liquid phase reactor is depicted in Fig. 17.3. The liquid fissionable material is held by centrifugal forces in the walls of cylinder from the porous material. The cylinder indicated with the work of engine is rotated with the aid of the special system. Working body passes through the pores of cylinder, cooling it, and then through a layer of the liquid fissionable material it is heated to the high temperature.

The liquid fissionable material can be used in the pure form or in the form of mixture with the substance, which possesses very high melting point. For example, as the liquid fissionable material can serve carbide of uranium UC_2 in molten form ($T_{пл} = 2575°K$), and also the mixture of carbide of uranium with carbide of tungsten or zirconium carbide; this mixture can be used in the reactor at a temperature about 4675°K.

A number of very promising YaRD includes the engines with the gas-phase reactor.

Are most investigated theoretically and experimentally two types of the gas-phase reactors:

a) with the rotation of fissionable material (in particular, with the utilization of the rotating magnetic field) and the radial motion of working medium/propellant from the periphery to the center; this type of reactor in many respects is similar to liquid phase reactor examined above;

b) coaxial type reactor (Fig. 17.4).

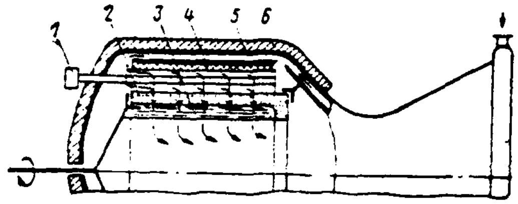


Fig. 17.3. Chamber/camera of YaRD with the liquid phase reactor: 1 - control rod with the drive; 2 - rotating cylinder with porous diaphragm; 3 - liquid fissionable material; 4 - reflector; 5 - chamber wall; 6 - protective shield.

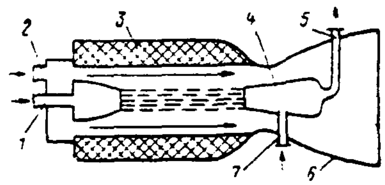


Fig. 17.4. Diagram of chamber/camera of YaRD with coaxial type gas-phase reactor: 1 - conduit/manifold of delivery of liquid fissionable material from separator; 2 - conduit/manifold of delivery of working medium/propellant from tank; 3 - retarder-reflector; 4 - air intake from porous material; 5 - conduit/manifold of branch/removal of mixture of fissionable material and working medium/propellant into separator; 6 - nozzle; 7 - conduit/manifold of delivery of working medium/propellant from tank.

Page 317.

In coaxial type reactor fissionable material (for example Pu^{239} or U^{235}) it is supplied in the liquid state. Directly in the reactor as a result of the course of nuclear reactions fissionable material is heated to the high temperature, it changes into the gaseous state and it moves along the axis/axle of reactor, being mixed only to the low degree with the working medium/propellant which moves coaxially with the flow of fissionable material (outside it). The stream of fissionable material recovers by the air intake, established/installed at the nozzle entry.

In the air intake the fissionable material is cooled and is condensed with its mixing with the cold working medium/propellant, which enters from the tank, it heads for the separator, and from it again into the reactor, etc.

In connection with the high temperatures of working medium/propellant in YaRD with the liquid phase and gas-phase reactor it is necessary to use the porous cooling of the walls of reactor.

§17.6. Difficulties of designing YaRD.

In spite of the fact that works on creation YaRD are conducted intensively, the engines, suitable for utilization on the rockets or KA, it is not yet developed. For example, the flight tests YaRD with

the solid-phase reactor and by thrust 340 kn [~~34 T] are expected in the USA not previously sulfurdi the 70's.

Significant difficulties during the creation YaRD produces the need for protection from the γ-rays and the fast neutron flux, which appear upon decay of the nuclei or fissionable material. The large mass of this protection for YaRD, utilized in the rocket vehicle, is not admitted; at the same time to its effectiveness, especially for the engines of the manned spacecraft, are presented quite high requirements.

Operation of YaRD is significantly more complex than the operation of other types of rocket engines. The presence of the service personnel about YaRD is admissible only before his first testing, after which all operations/processes, including taking apart from the stand and transport from the stand, must be performed remotely/distance.

In order to exclude the contamination of the atmosphere and large earth sections by radioactive decay products with the emergencies of rockets with YaRD, especially at the moment of start, such engines can be used only in the upper booster stages, and also in KA and spacecraft.

During the develcpment of YaRD it is necessary to solve number of complex problems which include: provision of heat transfer in nuclear reactor and reliable cooling of its elements/cells, the selection of materials of active region/core, the exception/elimination of inadmissibly high thermal stresses, etc.

Construction/design of YaRD to a considerable degree becomes complicated by the presence of the automatic systems of control of the mode/conditions of the work of reactor and engine as a whole, moreover the series of elements of the systems indicated, placed within the reactor, they work under conditions for acute irradiation.

Page 218.

Chapter XVIII.

ELECTRICAL ROCKET ENGINES.

The electrical rocket engines include by thermal type ERD, the electrostatic and electromagnetic engines (see §1.2, 1.6, 3.3). The distinctive special feature/peculiarity of engine installations with ERD is their large specific mass due to the presence of the source of electrical energy. Therefore the thrust-weight ratio [1] of rocket vehicle with ERD (10^{-1}-10^{-4}) is substantially less than vehicle with ZhRD or YaRD with the reactor of division (1-10).

FOOTNOTE [1]. The thrust-weight ratio of rocket vehicle call the ratio of thrust its engine installations to the weight of vehicle at the level of sea. ENDFOOTNOTE.

This gives rise to the low accelerations of vehicle which can ensure with ERD (usually not more than 10^{-2}-10^{-5} g_3), and worthwhileness of their use/application only for the upper stages of space vehicles, and also for KA.

ERD can provide an optimal (in accordance with the requirement of flight) change in the thrust, the specific impulse and operating time over wide limits with small error. Thermal type ERD most expedient to use in the stabilization systems and orientation of satellites and KA. Electrostatic and electromagnetic RD can be applied as the sustainer engines of KA, intended for the long-distance flights (to Mars, Jupiter, Saturn and other planets).

For electrostatic and electromagnetic RD are characteristic the following special features/peculiarities.

1. Very high specific impulse (see Table 1.1) unattainable for RD, in which working body is accelerated by transformation of heat into kinetic energy, moreover it is reached at low (usually not more than 50-100 mm Hg) pressure of working medium/propellant and at its relatively low temperature.

2. High specific primary power (to 500 kW/N [~5000 kW/kg]) (see pg. 25).

3. Low thrust (0.1-500 MN [~0.01-50 g]) with the possibility of continuous (over the long term of up to several years) operation.

4. Impossibility of applying external flowing cooling due to low

of expenditure of working medium/propellant. Engines are cooled by radiation into outer space. Therefore, in spite of low heat fluxes from the plasma to the walls electrical RD, their cooling presents great difficulties.

Page 219.

By this is produced the need for use/application in such engines of the expensive fever and erosion-resistant materials.

In connection with the low expenditure of working medium/propellant in the engine installations with ERD it is possible to use not only the usual pressure feed systems, but also feed system with the utilization of forces of surface tension. For example, liquid cesium from the tank in ERD can enter under the effect/action of capillary forces on the porous nickel rod.

§18.1. Thermal type ERD.

A number of type electrical heat engines includes the resistor and arc-jet engines (see §1.6), and also engines with the exploding thin wires (see pg. 44).

For obtaining the high-temperature working medium/propellant

resistor of RD it is necessary to heat the surface of resistor to possibly the higher temperature. Therefore resistors are made from the high-temperature (strength) materials, in particular from the tungsten and the rhenium, which can work at a temperature 2000 and 2400°K respectively.

The basic parameters resistor of RD usually are located in the following range: $P_n = 10^{-3} \div 10^{-4}$ n [10^{-4}-$1)^{-5}$ kgf]; $I_{уд.п} = 1500 \div 8500$ N•s/kg [~~150-850 kg•s/kg]; the efficiency of engine - to 85c/c; specific heat output $N_{тепл.уд} = 0.8 - 5.0$ kW/N [~~8-50 kW/kg].

Resistor engines have sufficiently simple construction/design and possess the significant service life of work.

Electric arc RD create thrust as a result of the outflow of high-temperature plasma. The temperature of plasma in immediate proximity of the arc can reach 4•10⁴K, but mean temperature of entire mass of plasma does not usually exceed 5000°K. As a result of the high temperature of plasma the specific impulse of arc-jet engines has great value of all types thermal RD.

Are given below the ranges of the typical values of the basic parameters electric arc RD: $P_n = 0{,}001 \div 100$ n [~~0.01-10 kgf]; $I_{уд.п} = (7{,}5 \div 25) \cdot 10^3$ N•s/kg [~~750-2500 kg•s/kg]; the efficiency of engine

- 40-60o/o; $N_{\text{тепл.уд}}=6\div32$ kW/N [~~60-320 kW/kg].

It is possible to increase specific impulse, raising the temperature of plasma, but in this case it is decreased by the efficiency of engine it increases/grows $N_{\text{тепл.уд}}$, i.e. they are increased the necessary expenditures of the power: with the increase of temperature the conductivity of metals it stops the commensurable with the plasma conductivity, which causes an increase of the ohmic losses in the internal resistance of the source of electrical energy and in the supplying electrical chains.

The service life of the work of electric arc rocket engines is limited to the erosion of electrodes.

Page 320.

To a number of positive qualities electric arc RD, besides high specific impulse, should be related sufficient simplicity of construction/design.

§18.2. Electrostatic motors.

Exhaust jet electrostatic RD is the ionic flow or colloidal particles; the latter differ from ions in terms of the substantially

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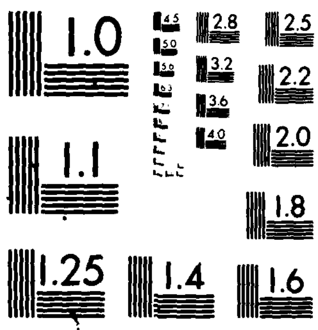

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smaller value of the ratio of particle charge e to its mass m; therefore electrostatic RD subdivide into the ionic ones and the colloidal ones.

Most intensively are developed/processed ionic RD. They are distinguished by the method of the ionization of working medium/propellant to the engines with the contact ionization (ions are formed with the contact of working medium/propellant with any warmed up surface) and the engines with the volumetric ionization, in which the ionization is provided by the high-frequency (to 200 MHz) field, which affects working the body, or as a result of its electron bombardment.

To the greatest degree are finished RD with the contact ionization, in which working medium/propellant is cesium (Fig. 18.1). It is supplied from tank 1 into vaporizer/evaporator 2 with the aid of the forces of surface tension. With the passage of vapors of cesium through ionizer 5 are formed the ions, which are accelerated/dispersed with the electrostatic field, which appears as a result of a potential difference between the ionizer and accelerating electrode 7. As ionizer usually serves disk made of the porous tungsten whose temperature by electric heater 4 brings itself to 1150-1350°K.

So that on housing of KA would not appear electric charge, which would lead to a reduction/descent in the engine thrust, exhaust jet of electric motors must be neutralized; for this purpose into the escaping ionic flow are introduced with the aid of the electrically heated grid (emitter) the electrons.

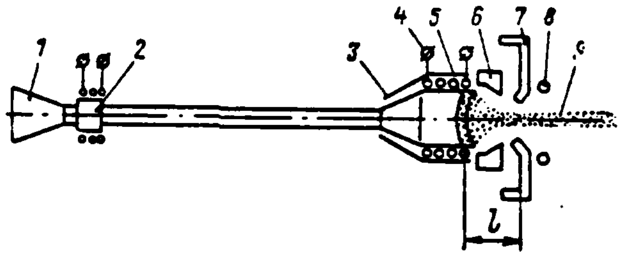


Fig. 18.1. Diagram electrostatic RD with the contact ionization (is working a body- cesium): 1 - tank with cesium; 2 - vaporizer/evaporator of cesium; 3 - heat shielding; 4 - electric heater of ionizer; 5 - ionizer made of the porous tungsten; 6 - focusing electrode; 7 - accelerating electrode; 8 - neutralizer; 9 - exhaust jet (mixture of ions and electrons); l - length of the section of the acceleration of ions.

Page 321.

Ion beam consists of the mutually repelling particles. Therefore, in order to avoid scattering of bundle and connected with this effect of the particles of the bundle on the engine components, in its construction/design is provided for focusing electrode 6.

The examination of the equations, which characterize electrostatic RD, shows that their specific impulse it is possible to increase, increasing relation m/e for the particles of the working

medium/propellant, but in this case simultaneously it is necessary to raise the strength of electrostatic field; however with its increase increases/grows the mass of the corresponding transformers of electrical energy of DU; furthermore, it can arise by the electric arc test/sample between accelerating electrode and ionizer.

Optimal relation m/e for the particles of the working medium/propellant will increase/grow in proportion to the perfection of the transformers of electrical energy of DU. At present relation m/e, close to the optimal, provide ions of such working medium/propellants as cesium and mercury.

Shortcoming electrostatic RD is the low thrust, per unit cross-sectional area f. If we designate this specific parameter through P_f, that

$$P_f = \frac{P}{f}.$$

Due to low values p_f the diameter and the mass of engine are sufficiently large.

§18.3. Electromagnetic engines.

The plasma of working medium/propellant electromagnetic RD is created by electric arc and it is accelerated/dispersed with external or proper magnetic field. In contrast to the electrostatic field the

magnetic field can accelerate electrically neutral plasma. Therefore for electromagnetic RD there is no need for both in the preliminary separation of positively and negatively charged/loaded particles and in the subsequent neutralization of exhaust jet. Due to the absence space electric charge electromagnetic RD can develop higher specific impulse than electrostatic. However, electromagnetic RD are inferior the electrostatic in value efficiencies (process of dispersing/accelerating the plasma by magnetic field is less effective than the process of dispersing/accelerating the ions by electrostatic field) and, furthermore, they have large mass.

The simplest example of electromagnetic continuous engine is the engine in which the plasma is accelerated/dispersed with the intersected electrical and magnetic fields. In this engine the channel over which moves the plasma, is rectangle, moreover its two opposite of wall are magnet poles (electrical or constants), and other two walls - by electrodes.

Page 322.

It is known that under the effect/action of electric field is differing charged/loaded of particle they move to opposite sides. But if we on the electric field superimpose perpendicular to it magnetic field, then on particles of both signs will act the forces, which have the identical direction, perpendicular to the direction of particle motion in the electric field.

In the pulse engine with the coaxial electrodes (Fig. 18.2) is not required external magnetic field. In this engine interacting with each other electrical and magnetic fields, and also plasma are created by one and the same electric arc. Chamber wall and arranged/located along its central axis/axle rod are coaxial electrodes. Working body (gas) is supplied into radial clearance between the electrodes. With the discharge of capacitor/condenser to the electrodes is created the electric field, as a result of which extremely rapidly the swelling current, causing the ionization of gas. As a result of the course of the current through the plasma appears magnetic field and under its effect/action the plasma is

compressed until sets in the equilibrium between the pressure of plasma and the pressure which is created by the effect/action of magnetic field. As a restlt of the process indicated the plasma is heated to the very high temperature and flows out behind the nozzle, creating thrust. The phenomenon the compressions of plasma with the course through it of electric current call pinch effect. After each impulse/momentum/pulse of thrust the capacitor/condenser is charged from the source of electrical energy how engine it is prepared for the creation of the following impulse/momentum/pulse.

The examination of the theoretical dependences, which characterize the dispersal/acceleration of plasma by magnetic field, shows that for obtaining the high efficiency of engine are necessary the high values of the coefficient of plasma conductivity and magnetic induction. As working medium/propellant electromagnetic RD can serve lithium, argon or hydrogen with the additions of potassium or sodium, etc.

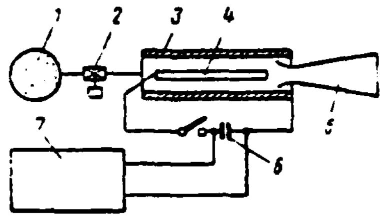


Fig. 18.2. Diagram pulse electromagnetic RD with the coaxial electrodes: 1 - tank/balloon with the working medium/propellant (gas): 2 - valve-reducer; 3 - chamber wall (external electrode); 4 - internal electrode; 5 - nozzle; 6 - capacitor/condenser; 7 - source of electrical energy.

Page 323.

§ 18.4. Sources of electrical energy (power unit).

Creation of effective DU with the electrical rocket engines in many respects is limited to the low values of the specific parameters (first of all, the specific power) of the sources of electrical energy, or of power units, to the explained in principle low efficiency of the power units (see pg. 25).

The classification of power units is shown in Fig. 18.3. According to the type of primary energy of power unit analogous with RD it is possible to subdivide into the chemical ones, the solar ones

and the nuclear ones (radioisotope and with the fission-type nuclear reactor).

The chemical power units include electrochemical storage batteries/accumulators (or chemical batteries) and fuel cells. In the electrochemical storage batteries/accumulators is expended/consumed the chemical energy, accumulated with their charging before the flight from the ground-based source of electrical energy.

In the fuel cells proceeds the process, reverse/inverse to the electrolysis of the water: to the porous electrodes, between which is located the electrolyte, are supplied gaseous oxygen and hydrogen (Fig. 18.4), as a result of electrochemical processes on the electrodes appears a potential difference and is formed water, which after certain cleaning/purification from the dissolved gas can be used by cosmonauts for the drinking.

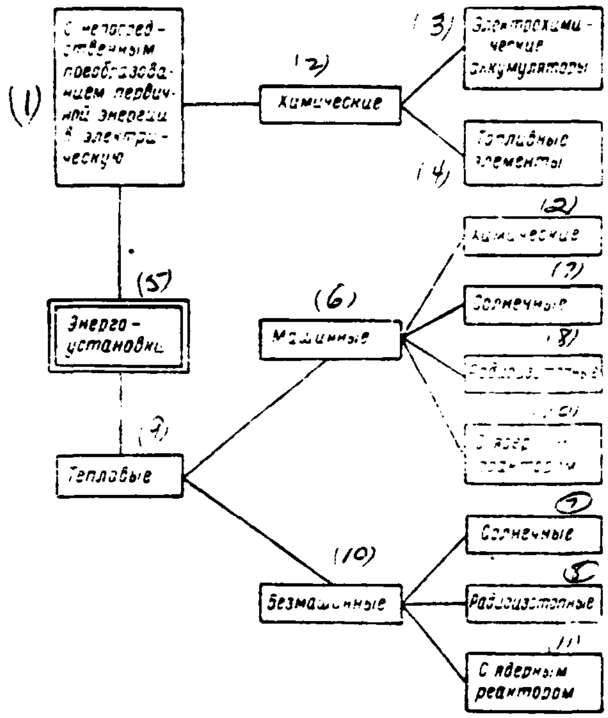


Fig. 18.3. Classificaticn of tne pcwer units of rocket vehicles.

Key: (1). With the direct transformation cf primary energy into the electrical. (2). Chemical. (3). Electrochemical storage batteries/accumulators. (4). Fuel cells. (5). Pcwer units. (6). Machine. (7). Solar. (8). Radioisotope. (9). Thermal. (10). With nuclear reactor. (11). With nuclear reactcr.

Page 324.

In the solar power units (cr sclar batteries) are used the

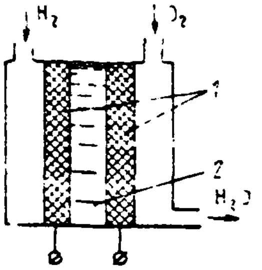
photoelectric transformers, which produce electrical energy with their illumination by solar rays/beams. If the service life of chemical power units is limited (usually it does not exceed several weeks), then solar batteries can work long time (of up to several years).

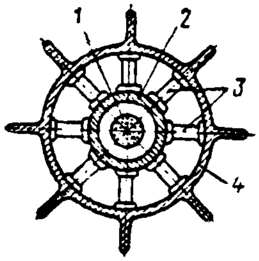
Thermal power units are subdivided into the machine ones (turbogenerator) and the machine-free ones.

In the machine-free power units thermal energy directly is converted into the electrical. Fig. 18.5 shows the diagram of radioisotope power unit with the thermionic converter, which are the battery of the thermocouples whose one weld is placed into the zone of the liberation of heat - into the capsule with the radioisotope, and another is connected with the low-temperature zone - radiator. Radiator throws off excessive heat into outer space. This power unit also can work several years.

In the machine power units (Fig. 18.6) are an electric generator and a closed duct/contour of the working medium/propellant, into which enter the turbine, radiator-capacitor/condenser, the pump of working medium/propellant and nuclear reactor. The vapors of working medium/propellant from nuclear reactor enter the turbine, giving its and connected with it electric generator in the rotation, and then

into capacitor-radiator, in which the vapors of working medium/propellant are condensed. The liquified working body is pumped into nuclear reactor, etc. Machine power units especially with nuclear reactor) are capable of creating large power during the long time (of up to several years).





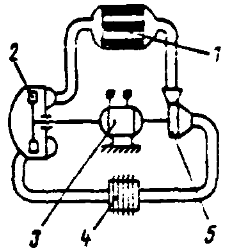


Fig. 18.4. The diagram of the fuel cell: 1 - porous electrodes; 2 - electrolyte.

Fig. 18.5. Diagram of radioisotope power unit with thermionic converter: 1 - capsule with radicisotope; 2 - internal (hot) shell; 3 - elements/cells of semiconductor thermionic converters; 4 - outer covering (radiator).

Fig. 18.6. Diagram of machine power unit: 1 - nuclear reactor; 2 - turbine; 3 - electric generator; 4 - radiator-condenser; 5 - pump of working medium/propellant.

Page 325.

Appendix 1.

Most frequently used units of the measurements of the system of SI.

Unit of mass - kilogram (kg.) - the mass of one liter (l) of the distilled water at a temperature 277°.15K [4°C] (1 l=1.000028 dm^3).

Unit of force - newton (N) - the force, which communicates to the body with a mass of 1 kg. the acceleration, equal to 1 m/s^2, i.e., 1 N=1 kg•m/s^2.

Unit of pressure - bar - the evenly distributed pressure at which on 1 m^2 acts normal to surface the force, equal to 10^5 N, i.e., 1 bar=10^5 N/m^2.

Unit of work - joule (J) - the work which accomplishes the constant force, equal to 1 N, with the displacement of the point of the application of this force on 1 m over its direction, i.e., 1 J=1 N•m.

Unit of power - watt (W) - the power, at which in 1 s is made the work, equal to 1 joule, i.e., 1 W=1 J/s.

Unit of specific heat - 1 J/kg - quantity of heat, which is contained in 1 kg. of mass.

Unit of heat flux - watt (W) - the heat flux with which through certain area 1 s passes the quantity of heat, equal to 1 joule.

Unit of heat transfer rata (neat-flux density) - W/m² - heat transfer rate, with which through the area, equal tc 1 m², is transmitted the heat output, equal to 1 W.

Appendix 2. Relationships/ratios between most frequently used unity of the measurements cf different systems.

1 н=0,1019716 кГ	1 кГ=9,80665 н
1 бар=1,019716 кГ/см²	1 кГ/см²=0,980665 бар
1 бар=0,986923 физ. атм	1 физ. атм=1,01325 бар
1 бар=750,062 мм рт. ст.	1 мм рт. ст.=1,33322·10⁻³ бар
1 н·сек/кг=0,101976 кГ·сек/кг	1 кГ·сек/кг=9,80665 н·сек/кг
1 н·сек/м³=101,9716 кГ·сек/л	1 кГ·сек/л= =9,80665·10⁻³ н·сек/м³
1 дж=0,1019716 кГ·м	
1 дж=0,238844 кал	1 кГ·м=9,80665 дж
1 кдж/кг=0,238844 ккал/кг	1 кал=4,1868 дж
1 вт=0,1019716 кГ·м/сек	1 ккал/кг=4,1868 кдж/кг
1 квт=1,35962 л. с.	1 кГ·м/сек=9,80665 вт
1 вт=0,238844 кал/сек	1 л. с.=0,735499 квт
1 вт/м²=0,859845 ккал/(ч·м²)	1 кал/сек=4,1868 дж
	1 ккал/(ч·м²)=1,1630 вт/м²

Key: (1) N. (2) kgf. (3) bar. (4) phys. atm. (5) mm Hg. (6) N•s/kg. (7) kg•s/kg. (8) kg•s/L. (9) joule. (10) N•s/m³. (11) cal. (12) kJ/kg. (13) kcal/kg. (14) W. (15) hp. (16) cal/s. (17) h•m². (18) hp.

Page 326.

Appendix 3. Table of values.

$\bar{f}_c = f(\varepsilon_c, n_p)$

ε_c \ n_p	1,08	1,09	1,10	1,11	1,12	1,13
20	4,3145	4,2458	4,1787	4,1138	4,0537	3,9941
40	7,4812	7,3225	7,1807	7,0372	6,9020	6,7731
60	10,405	10,164	9,9370	9,7148	9,5040	9,2997
80	13,193	12,860	12,548	12,244	11,959	11,685
100	15,875	15,458	15,062	14,675	14,312	13,961
150	22,320	21,677	21,049	20,457	19,906	19,372
200	28,476	27,594	26,779	25,970	25,213	24,486
250	34,442	33,320	32,263	31,261	30,308	29,394
300	40,270	38,902	37,635	36,445	35,266	34,180
400	51,450	49,724	48,003	46,362	44,769	43,306
500	62,359	60,108	57,880	55,840	53,921	52,084
750	88,507	85,077	81,783	78,682	75,758	72,937
1000	113,79	109,05	104,66	100,45	96,564	92,770
1500	162,26	154,95	148,32	142,12	136,16	130,54
2000	208,54	199,10	190,06	181,70	173,66	166,24
2500	253,58	241,53	230,36	219,84	210,14	200,74
3000	297,83	283,49	269,90	257,21	245,35	234,30
3500	340,48	323,55	307,84	293,02	279,23	266,30
4000	383,43	364,06	346,27	329,36	313,75	298,98
4500	425,91	404,22	384,04	364,94	347,36	330,60

ε_c \ n_p	1,14	1,15	1,16	1,17	1,18	1,19	1,20
20	3,9374	3,8807	3,8272	3,7738	3,7225	3,6726	3,6248
40	6,6462	6,5249	6,4097	6,2929	6,1830	6,0771	5,9756
60	9,1066	8,9200	8,7359	8,5595	8,3929	8,2280	8,0708
80	11,417	11,162	10,916	10,679	10,450	10,231	10,019
100	13,625	13,302	12,990	12,689	12,401	12,126	11,862
150	18,858	18,370	17,900	17,445	17,006	16,588	16,183
200	23,804	23,139	22,505	21,897	21,307	20,755	20,216
250	28,530	27,699	26,893	26,137	25,405	24,713	24,042
300	33,116	32,103	31,145	30,220	29,347	28,511	27,708

Page 327.

Continuation.

n_c \ n_p	1,14	1,15	1,16	1,17	1,18	1,19	1,20
400	41,878	40,546	39,257	38,046	36,865	35,574	[illegible]
500	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
750	[illegible]	67,787	[illegible]	[illegible]	60,969	[illegible]	[illegible]
1000	89,288	85,902	82,764	79,737	76,849	[illegible]	[illegible]
1500	125,28	120,22	[illegible]	[illegible]	106,78	[illegible]	[illegible]
2000	159,31	152,60	146,33	140,41	134,80	129,52	124,43
2500	191,89	183,74	175,96	168,50	161,55	155,04	[illegible]
3000	223,73	213,89	204,55	195,77	187,42	179,63	[illegible]
3500	254,04	242,68	[illegible]	[illegible]	212,06	[illegible]	[illegible]
4000	285,11	271,86	259,64	248,10	[illegible]	226,89	[illegible]
4500	315,07	300,33	286,50	273,50	261,18	249,67	[illegible]

n_c \ n_p	1,21	1,22	1,23	1,24	1,25
20	3,5783	3,5325	3,4900	3,4469	3,4060
40	5,8761	5,7803	5,6886	5,5977	5,5107
60	7,9208	7,7741	7,6317	7,4948	7,3634
80	9,8166	9,6176	9,4271	9,2438	9,0667
100	11,606	11,360	11,122	10,893	10,669
150	15,800	15,425	15,075	14,731	14,400
200	19,707	19,207	18,745	18,285	17,847
250	23,403	22,778	22,196	21,630	21,087
300	26,954	26,211	25,515	24,837	24,192
400	33,706	32,724	31,795	30,911	30,050
500	40,100	38,902	37,751	36,631	35,587
750	55,075	53,308	51,596	50,000	48,446
1000	69,114	66,753	64,529	62,386	60,364
1500	95,328	91,820	88,593	85,463	82,524
2000	119,72	115,21	110,88	106,86	102,99
2500	142,90	137,30	132,09	127,08	122,28
3000	165,34	158,67	152,46	146,52	140,87
3500	186,57	178,89	171,63	164,84	158,41
4000	207,95	199,34	191,14	183,36	176,01
4500	228,57	218,89	207,29	201,23	192,95

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Page 328.

Appendix 4. Table of values.

$K_p = f(\varepsilon_c, n_p)$

ε_c \ n_p	1,08	1,09	1,10	1,11	1,12	1,13
[illegible]	[illegible]	[illegible]	[illegible]	1,6409	1,6345	[illegible]
[illegible]	[illegible]	1,7625	[illegible]	1,7433	1,7342	[illegible]
[illegible]	[illegible]	1,8200	[illegible]	1,7974	1,7869	[illegible]
80	1,8724	1,8590	1,8460	1,8338	1,8221	[illegible]
100	1,9023	1,8880	1,8741	1,8608	1,8484	[illegible]
150	1,9548	1,9382	1,9230	1,9082	1,8941	1,8804
200	1,9901	1,9725	1,9559	1,9402	1,9246	1,9096
250	2,0168	1,9981	1,9806	1,9634	1,9475	1,9316
300	2,0380	2,0186	[illegible]	1,9823	1,9655	1,9490
400	2,0702	2,0496	2,0297	2,0111	1,9928	1,9755
500	2,0945	2,0726	2,0517	2,0321	2,0131	1,9951
750	2,1371	2,1134	2,0908	2,0691	2,0488	2,0287
1000	2,1662	2,1409	2,1172	2,0945	2,0728	2,0517
1500	2,2055	2,1784	2,1528	2,1281	2,1049	2,0825
2000	2,2322	2,2040	2,1764	2,1509	2,1264	2,1032
2500	2,2524	2,2226	2,1947	2,1681	2,1430	2,1185
3000	2,2686	2,2381	2,2092	2,1816	2,1557	2,1309
3500	2,2816	2,2504	2,2210	2,1928	2,1665	2,1408
4000	2,2931	2,2612	2,2313	2,2026	2,1754	2,1493
4500	2,3030	2,2709	2,2399	2,2108	2,1832	2,1567

ε_c \ n_p	1,14	1,15	1,16	1,17	1,18	1,19	1,20
20	1,6223	1,6164	1,6107	1,6050	1,5994	1,5947	1,5897
40	1,7167	1,7088	1,7007	1,6930	1,6857	1,6787	1,6717
60	1,7670	1,7571	1,7482	1,7392	1,7303	1,7223	1,7144
80	1,8000	1,7894	1,7796	1,7695	1,7599	1,7513	1,7423
100	1,8246	1,8132	1,8026	1,7920	1,7817	1,7724	1,7628
150	1,8675	1,8547	1,8427	1,8309	1,8195	1,8089	1,7982
200	1,8959	1,8822	1,8691	1,8564	1,8440	1,8330	1,8215
250	1,9169	1,9025	1,8889	1,8754	1,8625	1,8506	1,8386
300	1,9338	1,9186	1,9042	1,8904	1,8769	1,8644	1,8519

Page 329.

Continuation.

ε_c \ n_p	1,14	1,15	1,16	1,17	1,18	1,19	1,20
400	1,9591	1,9430	1,9277	1,9128	1,8986	1,885[illegible]	1,8721
500	1,9778	1,9610	1,9450	1,9294	1,9146	[illegible]	[illegible]
750	2,0103	1,9921	1,9749	1,9559	1,9421	[illegible]	1,9123
1000	2,0321	2,0130	1,9950	1,9773	1,9602	1,9447	1,9292
1500	2,0615	2,0408	2,0216	2,0028	1,9848	[illegible]	1,9513
2000	2,06[illegible]	2,0596	2,0394	2,0196	2,0008	[illegible]	1,9661
2500	2,0957	2,0735	2,0527	2,0321	2,0129	1,9945	1,9769
3000	2,1072	2,0845	2,0631	2,0421	2,0223	2,0038	1,9855
3500	2,1167	2,0936	2,0717	2,0502	2,0300	2,01[illegible]	1,9926
4000	2,1249	2,1012	2,0789	2,0571	2,0366	2,0[illegible]73	1,9985
4500	2,1319	2,1079	2,0852	2,0632	2,0421	2,0227	2,0036

ε_c \ n_p	1,21	1,22	1,23	1,24	1,25
20	1,5848	1,5800	1,5759	1,5716	1,5676
40	1,6652	1,6587	1,6527	1,6467	1,6412
60	1,7067	1,6991	1,6922	1,6852	1,6788
80	1,7339	1,7258	1,7179	1,7106	1,7034
100	1,7540	1,7453	1,7370	1,7288	1,7213
150	1,7882	1,7784	1,7693	1,7603	1,7518
200	1,8107	1,8001	1,7903	1,7807	1,7715
250	1,8257	1,8161	1,8058	1,7956	1,7861
300	1,8401	1,8287	1,8178	1,8073	1,7973
400	1,8596	1,8475	1,8359	1,8248	1,8143
500	1,8738	1,8612	1,8492	1,8377	1,8266
750	1,8965	1,8846	1,8718	1,8594	1,8476
1000	1,9116	1,9001	1,8868	1,8736	1,8614
1500	1,9358	1,9205	1,9064	1,8924	1,8794
2000	1,9525	1,9341	1,9191	1,9047	1,8911
2500	1,9602	1,9439	1,9286	1,9138	1,8998
3000	1,9655	1,9517	1,9361	1,9208	1,9067
3500	1,9750	1,9580	1,9421	1,9266	1,9121
4000	1,9778	1,9634	1,9473	1,9315	1,9167
4500	1,9854	1,9680	1,9516	1,9357	1,9207

Pages 330-332.

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Page 331.

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8